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T-38A CATEGORY II STABILITY AND CONTROL TESTS

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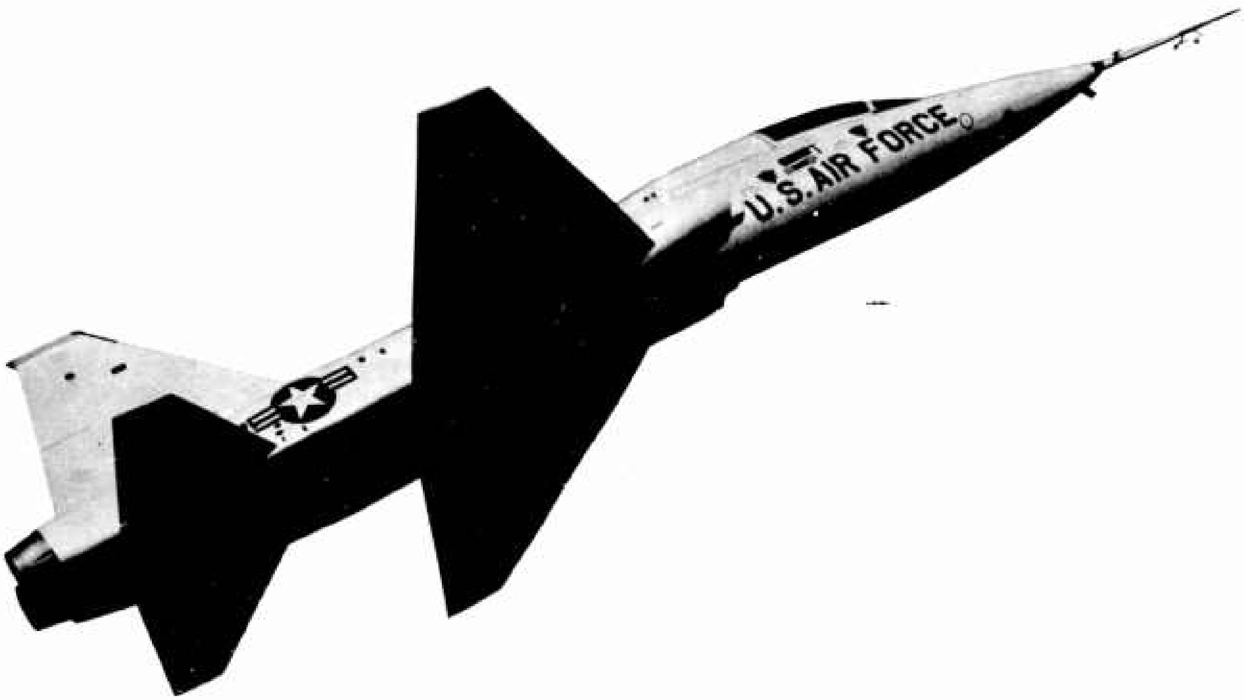

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ABSTRACT

This report presents the results of the Air Force Flight Test Center Category II stability and control tests of the T-38A aircraft equipped with YJ85-GE-5 afterburning engines. The AFFTC test program was conducted under the integrated contractor/Air Force Category System and consisted of 17 flights totaling 13.8 hours of flying time. Approximately 9.5 hours of contractor obtained data was also utilized in the preparation of this report. T-38A and YT-38 aircraft equipped with YJ85-GE-5 and YJ85-GE-1 engines, respectively, were utilized during the test program.

The T-38 airplane, Air Force System 420L, is a two place tandem supersonic trainer manufactured by the Norair Division of the Northrop Corporation. The mission of the airplane is to accomplish all phases of basic pilot training, including; day and night transition, formation, navigation, instrument and acrobatic flying. The aircraft has a full powered irreversible flight control system which includes a conventional artificial feel system.

The T-38 aircraft is an excellent trainer for pilot transition into high performance jet aircraft including present Century Series fighter aircraft. Its simplicity is conducive to transition; its performance and handling characteristics allow Century Series simulation and development of correct pilot techniques. Visibility from the front seat and the rear seat is excellent during all phases of flight. The rear seat (instructor's position) affords complete control over most of the normal and emergency procedures.

The stability and control characteristics of the aircraft are satisfactory throughout the flight envelope. Most of the undesirable features noted during the Category I tests have been improved or corrected. The longitudinal control response is sensitive at high speeds and/or Mach numbers and slow at airspeeds below 220 knots IAS, but it is the best balance available from the simple non-q-biased flight control system. The aircraft is safe to fly with or without stability augmentation (pitch and yaw dampers) although the damping of longitudinal disturbances with the pitch damper inoperative is low and

military specification requirements are not met in almost all flight conditions at altitudes below 30,000 feet. The damper systems provide adequate damping at all airspeeds and slightly excessive damping longitudinally at low airspeeds. All inflight directional, lateral-directional, and lateral characteristics are satisfactory. There is a wide pre-stall buffet boundary which starts at airspeeds as high as 190 knots IAS in the cruise configuration and 170 knots in the power approach configuration, and continues to the minimum airspeed of about 115 knots. The buffet is unsatisfactory as a stall warning; however, lateral roll-off near the stall and lateral oscillations which occur just prior to stall provide a satisfactory stall warning.

The low nose wheel steering response near neutral requires excessive rudder pedal movements while taxiing and occasionally causes a low frequency divergence in directional oscillations on take-off due to over-control. This divergence can be stopped by releasing the nose wheel steering. There are a number of discrepancies in the cockpit layout and controls which should be corrected. None of these are unacceptable if adequately discussed in the Pilot's Handbook.

This report has been
reviewed and approved

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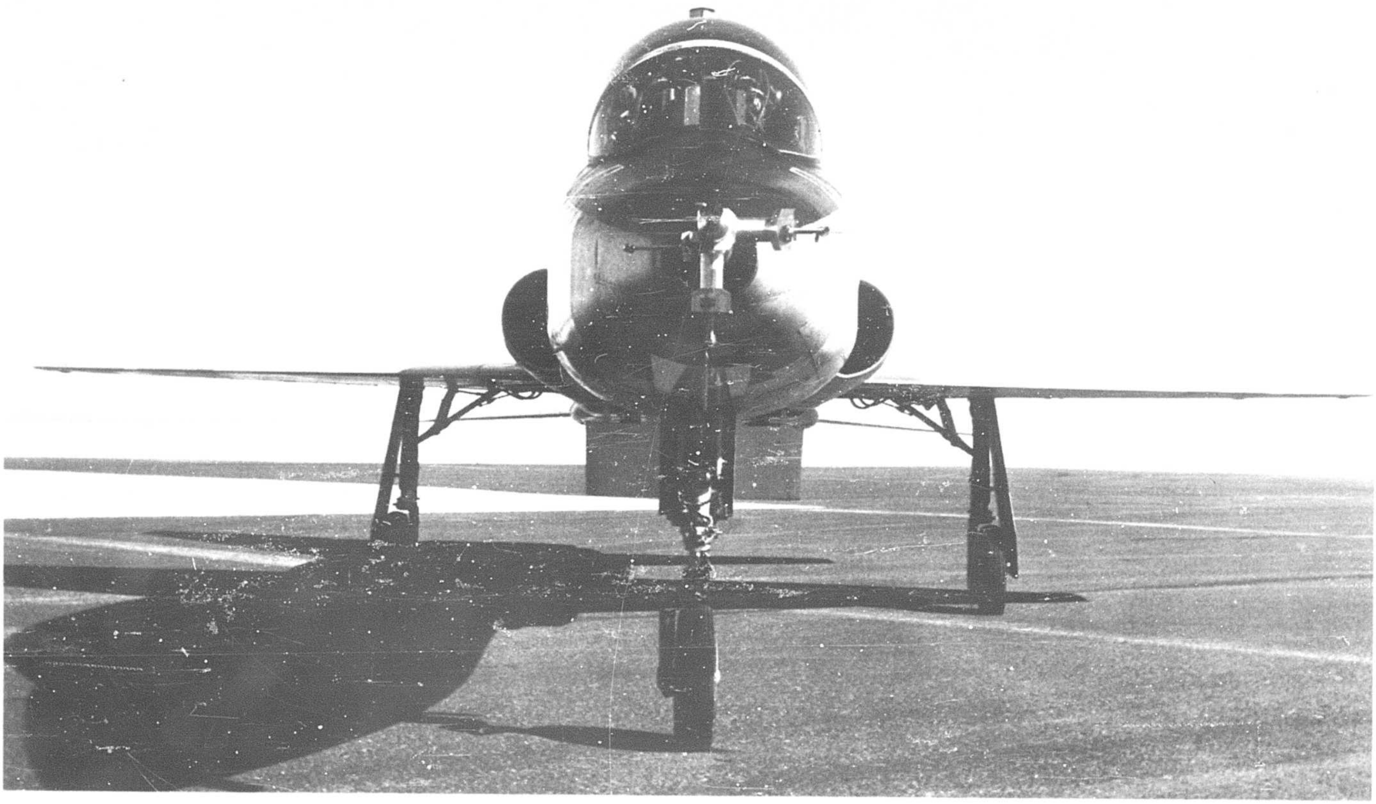
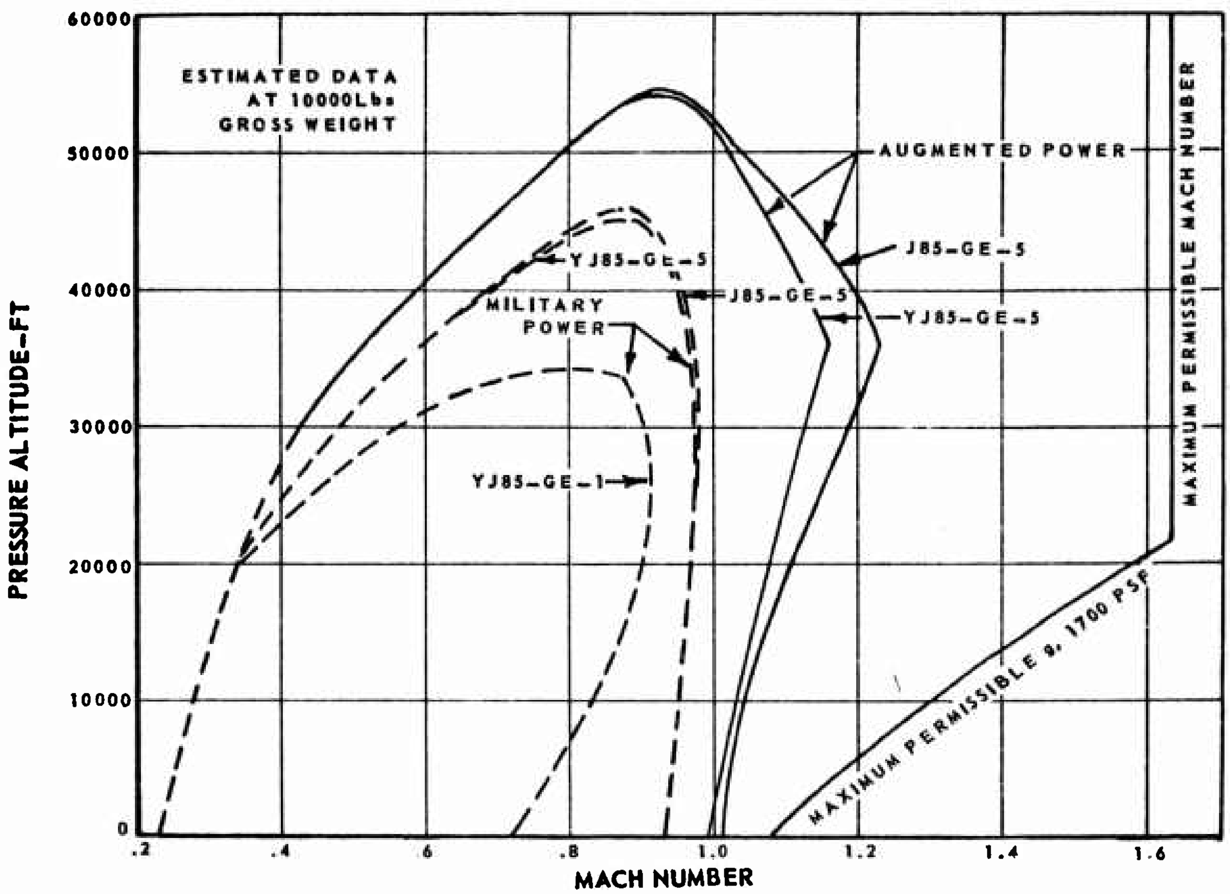


INTRODUCTION

This report presents the results of the Air Force Flight Test Center T-38A Category II stability and control tests. The tests were performed on contractor maintained aircraft which were being used concurrently by the contractor in Category I tests. The AFFTC Category II tests were conducted to obtain quantitative data on the stability and control and handling characteristics of the T-38 airplane in the flight envelope not tested during the Category I program, evaluate changes resulting from recommendations made during the Category I tests, and more thoroughly investigate some of the areas previously tested. The test data was obtained using YT-38 and T-38A aircraft equipped with YJ85-GE-1 engines and YJ85-GE-5 afterburning engines, respectively. The Air Force Flight Test Center program was flown at Edwards AFB, California, from 22 February to 16 December 1960 and consisted of 17 flights totaling 13.8 hours. Approximately 9.5 hours of contractor obtained data was also utilized in the preparation of this report. All contractor flights were monitored by the AFFTC surveillance Team. In addition to the Category II test flights, AFFTC pilots participated in the contractor's Category I program which was being conducted concurrently. This made it possible to evaluate various modifications as they were incorporated in the aircraft. Pilot participation was such that contractor and Air Force pilots participated in both the AFFTC and Norair test programs.

The T-38 airplane, Air Force Weapon System 420L, is a two place tandem trainer manufactured by the Norair Division of the Northrop Corporation. The mission of the airplane is to accomplish all phases of basic pilot training, including; day and night transition, formation, navigation, instrument and acrobatic flying. The flight envelope of the airplane with the YJ85-GE-1, YJ85-GE-5 afterburning engines, and the J85-GE-5 afterburning engines is presented in Figure A. It was not practical to obtain flight data above Mach 1.1 in level stabilized flight with the prototype -5 engine because acceleration above 1.1 Mach number is slow and consumes considerable fuel. The two

FIGURE A
T-38A ALTITUDE-MACH NUMBER
FLIGHT ENVELOPE



test aircraft had an average basic weight of 8,100 pounds with an internal fuel capacity of 582 gallons or 3,780 pounds (gross weight = 12,010 pounds). The present anticipated weight of a fully loaded (2 pilots and 3,800 pounds of fuel) production trainer is 11,550 pounds. With two pilots and a full load of fuel, the center of gravity position is about 20 percent MAC (the design center of gravity limits are 14 and 24 percent MAC, respectively). Both test aircraft had instrumentation installed in the rear cockpit. The external configuration of the test aircraft is the same as that presently planned for the production trainer except that the test aircraft was equipped with a test nose boom, a free air temperature bulb, and an externally mounted fuselage deflection camera and reference posts. All indicated airspeeds listed in this report are for the test nose boom.

The flight control system incorporated in the production aircraft has the same lateral and directional systems as evaluated during the AFFTC Category I tests; however, there were several modifications to the longitudinal control system. The

major differences between the prototype and the production longitudinal flight control system include:

1. Elimination of the stick damper

2. A revised automatic trim schedule during flap actuation

3. A new actuator servo valve arm and revised surface rates designed to give pushover response similar to pull-up response

4. A new feel spring

5. Revised electronic gain of the pitch and yaw dampers

Norair personnel assisted the Air Force Flight Test Center in the reduction of the stability and control test data. This data was immediately available to the contractor. Copies of the final plots contained in Appendix I of this report were sent to the T-38 Project Office, the Systems Design Principles Branch, WWDXH, and to the Norair Division of the Northrop Corporation on 12 April 1961.

The following configurations as listed in Military Specification MIL-F-8785(ASG), "Flying Qualities of Piloted Aircraft", are presented on the plots and referred to in the discussion:

Configuration	Symbol	Gear	Flaps	Power
Cruise	CR	Up	Up	Level flight
Power approach	PA	Down	Down 45°	Level flight
Take-off	TO	Down	Down 20°	Take-off
Landing	L	Down	Down 45°	Idle

TEST RESULTS

EXTERNAL INSPECTION

The preflight external inspection is easy to accomplish. The number of inspection items is small and their layout is such that no screw drivers or other tools are required.

COCKPIT EVALUATION

Individual ladders of excellent design provide easy entrance to each cockpit. Pull-out steps in the side of the fuselage provide a second means of entrance. Entrance via the pull-out steps is difficult but is adequate as a secondary system.

The cockpits are large and comfortable and are basically satisfactory; however, there are many items that should or could be improved. These items are classified as unsatisfactory (safety of flight involved), or needing improvement to provide a better product. With adequate pre-flight briefing and inflight caution, none of the items are unacceptable.

Unsatisfactory Items[1]:

*1. The friction detents which provide an idle stop for each throttle do not provide a positive stop and do not prevent an unintentional cut-off of all fuel to an engine when the throttle setting is reduced to idle. The possibility of an inadvertent stop cock and flame-out of one or both engines is a safety of flight hazard. A positive stop is required at the idle throttle position. (A 1)[2]

**2. The dimmed intensities of the landing gear position green lights, the J-4 compass warning light, and the fuel boost pump and crossfeed lights are all much too bright for night operations. The illumination in the cockpit reduces the visibility of objects and lights outside of the aircraft to a dangerous degree. The dimmed intensities of these lights should be reduced. (A 2)

*3. The fuel quantity gages are inaccurate near zero. Engines have flamed out in the air and on the ground with as much as 90 pounds of remaining fuel indicated on the gage for that engine. The fuel quantity indicating systems should be calibrated or modified to indicate the actual fuel remaining that is available to each engine. (A 3)

**4. During night flights, the reflections in the blast screen between the two cockpits are disconcerting to the pilot in the rear cockpit and can cause vertigo under some conditions. The wing and tail flasher lights of the aircraft are constantly reflected. Lights of other aircraft in formation occasionally appear as aircraft approaching from ahead. A sunset horizon or cloud formation or the lights of a city to the rear can cause serious disorientation and vertigo. This situation should be corrected by tilting or other modification of the blast shield. (A 4, C 2)

[1]* Indicated that corrective action has been initiated or approved

** Indicates that the item is under condideration by the T-38 Project Office

[2] Numbers indicated as (A1), etc., represent the corresponding recommendation numbers as tabulated in the Recommendation section of this report

*5. The present emergency landing gear extension system actuation requires a heavy pull force, held in an awkward position, for an extended period of time. The force required pattern during actuation is such that the maximum force required is just prior to full actuation. It is impossible to be sure of full actuation unless additional force is exerted against a positive stop at the end of the cable travel. The high force required is undesirable; the extended period of occupation of the left hand is unsatisfactory because it precludes throttle movement, flap actuation, airstart ignition actuation, etc., during the period. A single pull and release actuation is required. (A 5)

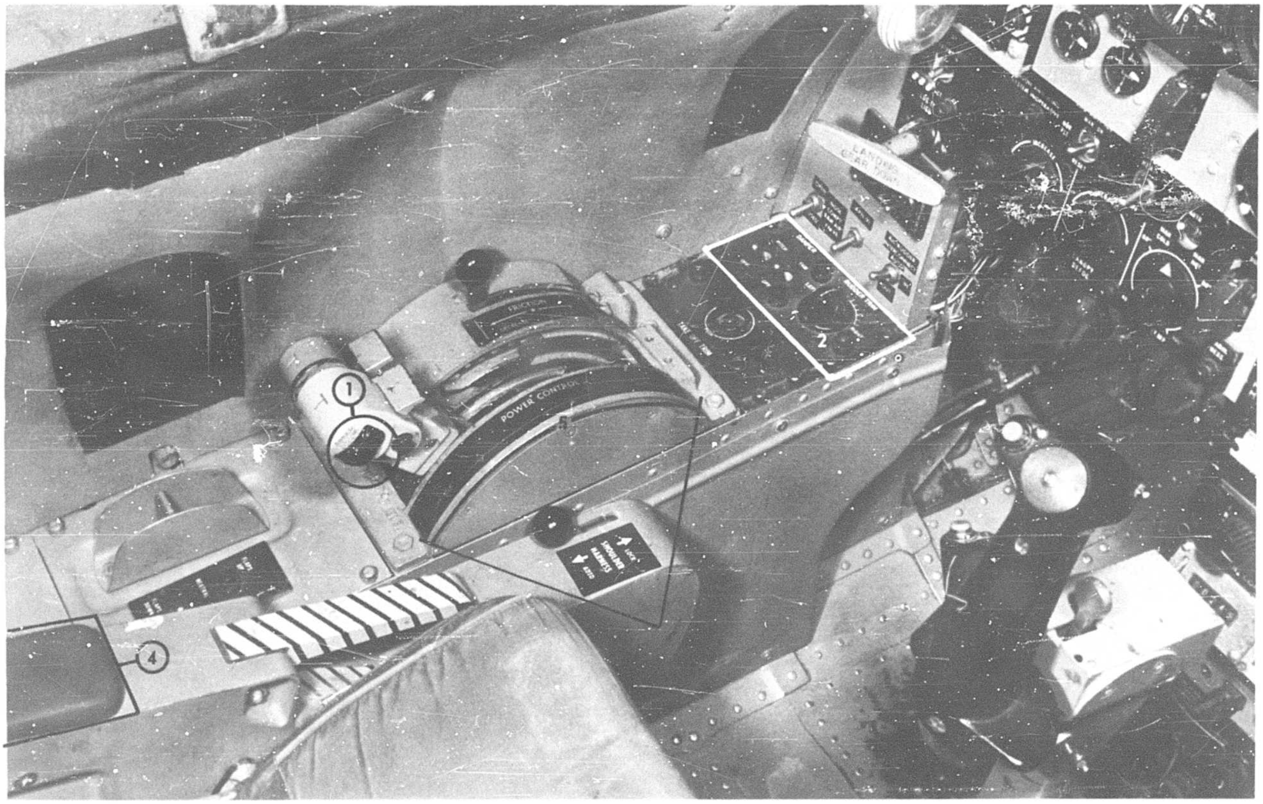
6. There is no standby compass in the rear cockpit. This precludes periodic comparison of J-4 compass readings to those of another system and provides no back-up compass for the instructor in the event of J-4 compass or electrical power failure. A standby compass should be installed in the rear cockpit. (A 6)

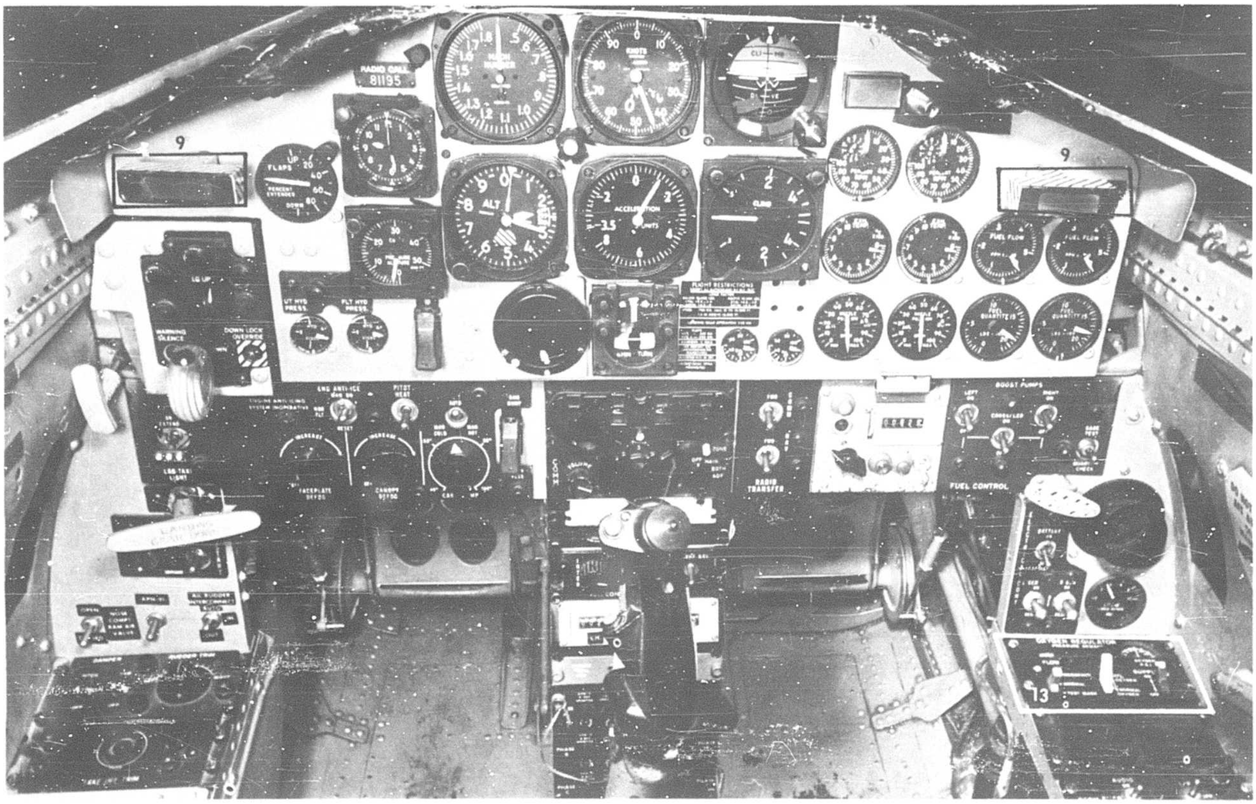
*7. The sidetone feedback of the interphone system is too low. The pilot transmitting is unable to hear his voice. This is a source of continuous annoyance and also requires a check on the receipt of critical instructions. The feedback volume should be adjusted to equal the output volume. (A 7)

Items Needing Improvement:

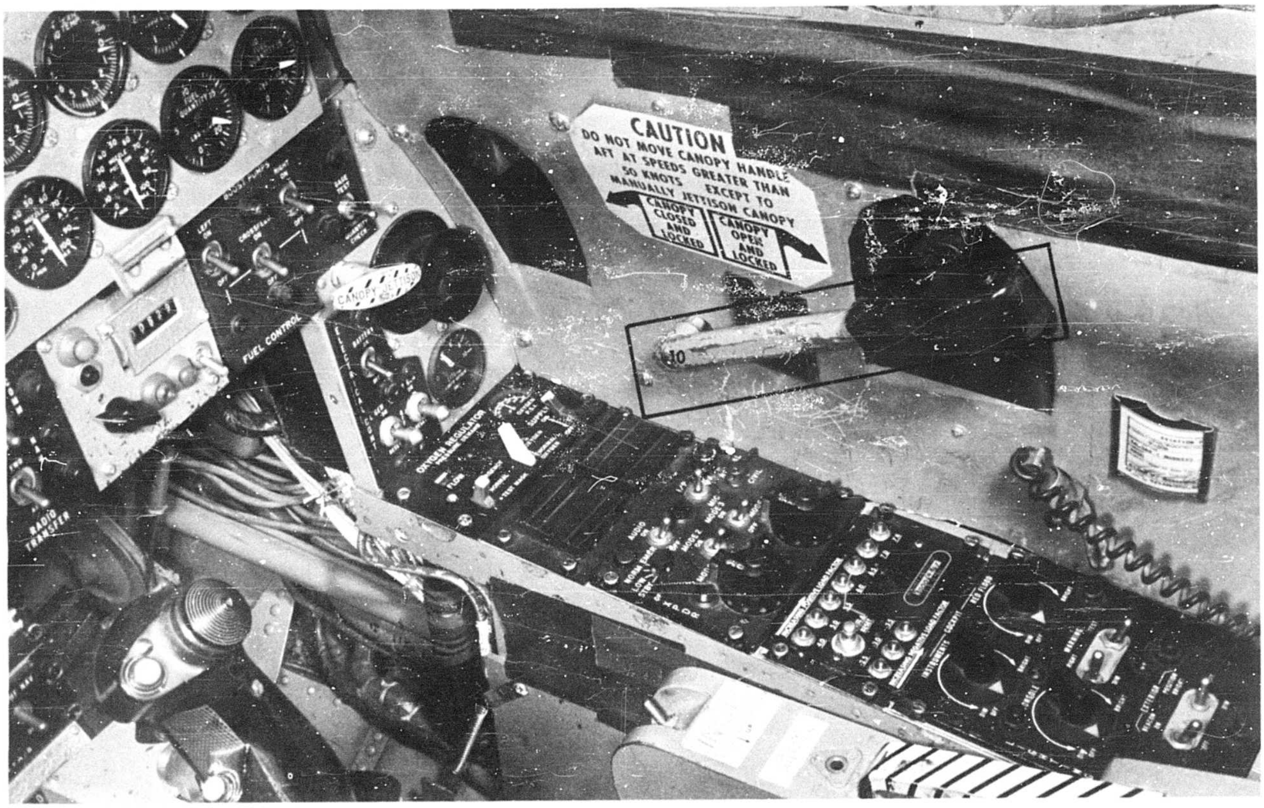
*1. The speed brake switch is too small and its positions too indefinite for positive feel and speed brake control. A larger switch with positive detents (in the front cockpit) should be installed. (B 1)

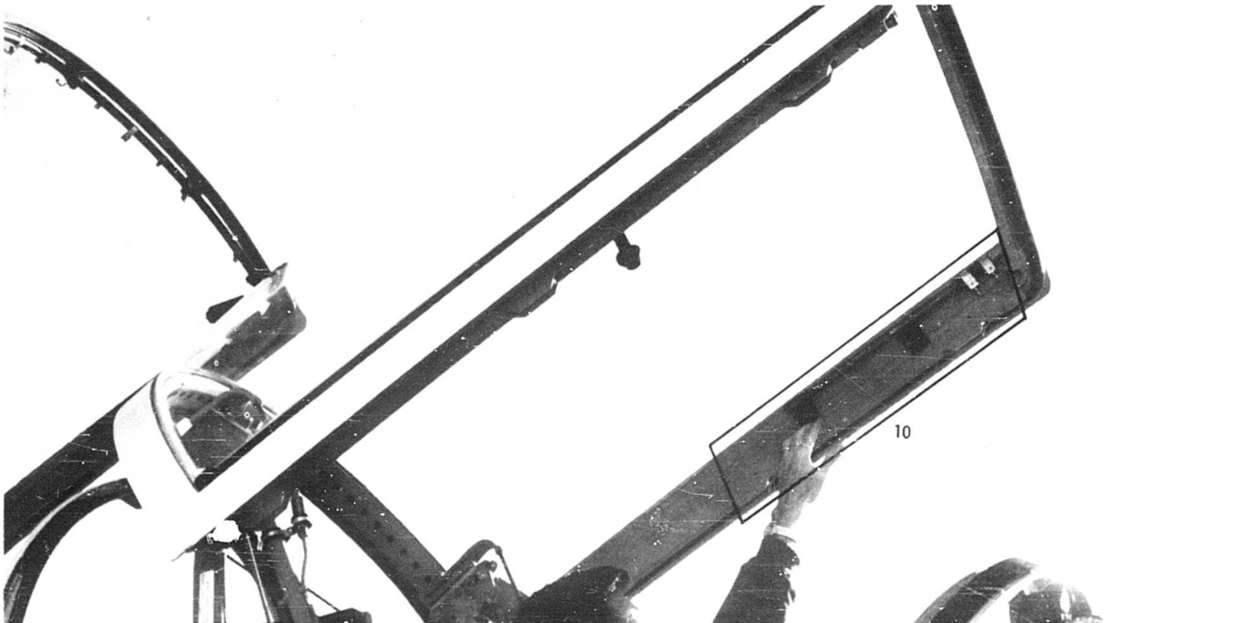
*2. The stability augmentation switches (front cockpit only) are located too far forward on the left console. The switches are beyond the reach of the pilot with the shoulder harness locked. The switches should be moved to aft of the throttle quadrant. (B 2)





COCKPIT ARRANGEMENT, T-38A S/N 58-1195, SHOWING NON-STANDARD COCKPIT WITH TEST INSTRUMENTATION AND EQUIPMENT. NUMBERS IN BOXES REFER TO CORRESPONDING ITEM LISTED UNDER "ITEMS NEEDING IMPROVEMENT."


CAUTION
DO NOT MOVE CANOPY HANDLE
AFT AT SPEEDS GREATER THAN
50 KNOTS EXCEPT TO
MANUALLY JETTISON CANOPY
CANOPY CLOSED AND LOCKED
CANOPY OPEN AND LOCKED
10
FUEL CONTROL
RADIO TRANSFER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*3. There is no indication in the rear cockpit of stability augmentation status. The instructor in the rear must ask the student in the front what the status is. It is necessary that the instructor know the current status, especially prior to certain types of demonstrations or maneuvers. An indicator system should be installed in the rear cockpit. (F 3)

**4. The pilot's seat and armrest arrangement provides no arm support except during ejection. During formation flights, while constantly holding and moving the engine power levers, the left arm becomes fatigued. A support would reduce arm fatigue and would serve as a reference and anchor point to allow wrist movement for throttle control. This is a design fault that should be corrected in later production of the seat. (B 4)

*5. The throttle sweep angles are too large, requiring an excessive angle change for a desired rpm change. This produces positions too far to the rear during power-on landing and instrument approaches, and extending beyond normal arm reach in the afterburner position. The total sweep angle should be reduced to about two thirds of the present angle and centered upon the present center of travel. (B 5)

6. The seat elevation rate is too slow but is merely an annoyance factor and can be tolerated. (B 6)

7. Longitudinal and lateral control trim rates are both too slow but are acceptable.

8. Landing light (*) and speed brake actuation rates are both too slow but are acceptable. (B 7)

*9. The removal of the fire warning and extinguisher handles from the instrument panel eliminated the fuel cut-off features resulting from their actuation. New fuel cut-off provisions should be installed. (B 8)

10. At normal taxi speed, or with a head wind blowing, the small amount of leverage available through the canopy operating lever is inadequate for easy closure of the canopy. Upon release for opening, the canopy occasionally blows to full open. Pilots generally use their left hand on the canopy itself (using the canopy defrost duct as a handle) to ease opening and closure forces. A handle should be installed on the canopy to provide a good grip. (B 9)

*11. The air conditioning system occasionally cycles and becomes extremely noisy following application of engine power. The system automatic controls should be improved. (B 10)

*12. A loud airhorn type noise suddenly develops at about Mach 1.3 during high speed dives. It can be reduced or eliminated by slowing the aircraft. (B 11, C 3)

**13. Visual monitoring of the warning light panel is partially obstructed by the pilot's right knee. Although the location of the master caution light on the main instrument panel alleviates the situation, tilting of the panel against the cockpit wall would eliminate the problem. (B 12)

ENGINE START

The T-38 requires the use of external air connected to each engine for ground starts. The installation of cross bleed air from the right engine for starting the left engine was tried as recommended in the Category I report. This arrangement proved unsatisfactory due to blasting of the ramp behind the aircraft with the exhaust from the right engine at the engine speed required for delivering sufficient starting air. The production aircraft will have a "Y" connection on the external air system which will allow starting of both engines from an external source without switching the external connection.

GROUND OPERATION

The YJ85-GE-5 and the J85-GE-5 engines are prohibited from continuous operation between 50 and 58 percent rpm due to a vibration problem. As a result, the pilot has a choice of taxiing too slowly at 49 percent, too fast at 59 percent, or continual power change from one side to the other of the prohibited band. This engine condition should be corrected. The nose wheel steering response to rudder pedal deflection is adequate at large deflections but too slow and limited for the small deflections about neutral normally used for taxi and take-off. The lack of response near neutral amounts to a dead band through which the pilot must continually operate while taxiing and which occasionally induces over control during take-off when the slow response prompts additional rudder selection. This eventually leads to using too much rudder. The aircraft can be taxied with brakes alone without nose wheel steering. (B 13 and 14)

Pedal forces for the braking action required to hold the aircraft at military power are excessive. (B 15)

Visibility, on the ground and in the air, is excellent from both cockpits except for the reflection problem in the rear cockpit previously mentioned.

TAKE-OFF AND CLIMB

Directional control during the take-off roll can be maintained with differential braking and rudder, or with nose wheel steering. Differential braking or nose wheel steering is required up to 50 knots IAS where the rudder becomes effective. There is an apparent dead band in the nose wheel steering around the neutral position. The response to normal taxi and take-off roll directional corrections is slow. These small directional corrections require large rudder deflections and can result in a divergent directional oscillation due to over-demand. Several modifications to the nose wheel steering system were evaluated but none of them corrected the dead band deficiency. (B 14, C 4)

Stabilizer effectiveness during take-off is adequate in that the aircraft can be rotated to the take-off attitude well below the stall speed. During augmented power take-off, with a normal take-off trim of zero degrees and a forward center of gravity of 15.7 percent MAC, the nose wheel can be raised at 120 knots IAS with full aft stick. The stick force at this forward cg is quite heavy (40 pounds) and the aircraft response to longitudinal stick movement is slow. Due to the high stick forces and the slow aircraft response, there is a tendency to overshoot the lift-off attitude and raise the nose too high so that the stick must be subsequently brought forward to its lift-off position. At the normal service cg position of 20 percent MAC the stick forces for lift-off are lighter and the aircraft response is better.

The aircraft tends to climb rapidly during the first few seconds after take-off and, due to the slow stabilator response, a definite push-over is required to accomplish a shallow initial climb out. The longitudinal trim change during flap retraction has been improved since the Category I test and the objectionable stick movement has been eliminated with the installation of the series trim system

LONGITUDINAL STABILITY AND CONTROL

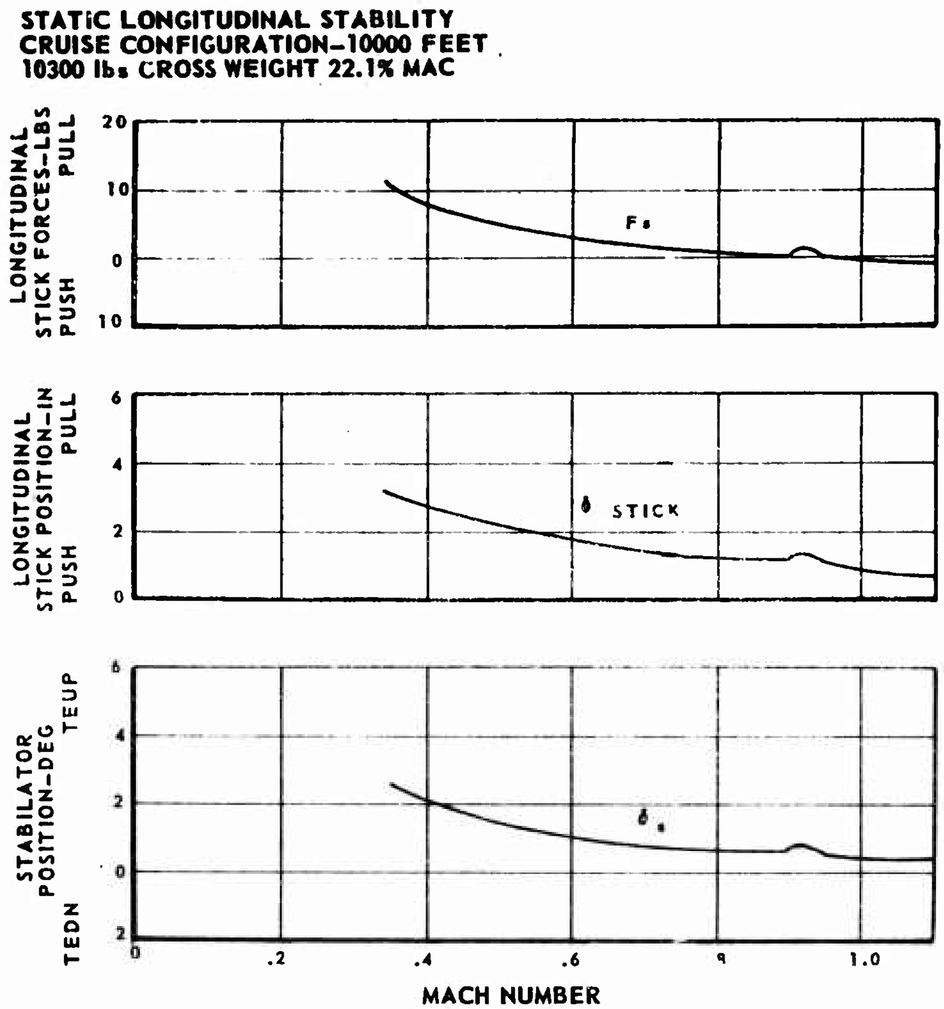
The production longitudinal flight control system, Norair MOD-50, was selected by AFFTC pilots during evaluation tests of several systems during July 1960. The stabilator response rate, although still slow at low speeds, was selected as a compromise between too high sensitivity at high airspeeds and too slow response at low airspeeds. The production longitudinal control system differs from the one tested in the AFFTC Category I tests as outlined in the following table:

Item	Category I	Category II	Remarks
Servo arm length	1.05 in	1.50 in	Changed linkage dynamics slightly.
Surface rate	25 deg/sec, T.E. up 18 deg/sec, T.E. down	26 deg/sec T.E. up and down	Changed aircraft pitch response characteristics.
Stick-surface gearing	----	----	Made stick travel versus surface travel relationship more linear around zero position.
Stick force	----	----	Feel spring modified to eliminate free play at trim.
Trim authority (gear up)	±6.3 deg	0 to 5 deg T.E. up	There was no need for the forward trim authority.
Stick damper	Various types	None	No damper required in production system.
Flap stabilator auto trim	----	----	Revised stabilator versus flap schedule and eliminated automatic repositioning of stick.

STATIC LONGITUDINAL STABILITY

The static longitudinal stability of the aircraft is satisfactory in all configurations throughout the speed and altitude envelope tested.

The modifications to the longitudinal control system eliminated the excessive free play about neutral. The longitudinal break-out force was reduced through improved rigging methods. The main longitudinal stability characteristics noted were very small stick force and position changes during slow acceleration and deceleration through wide speed ranges in the higher speed regime. During a deceleration at 10,000 feet from 1.1 Mach (626 knots IAS) to .7 Mach (390 knots), the total force and stick position changes were only 3 pounds and 0.7 inches, respectively. Further decrease in speed below 390 knots required more noticeable force and deflection inputs. There is a transonic trim change which occurs at about 0.9 Mach number; however, it is so mild that it is hardly perceptible, especially at low altitude. The aircraft is easily trimmed to desired conditions throughout its speed range. The present longitudinal trim system is satisfactory. The trim rate is slow but acceptable. The aircraft can be flown with trim alone. The pilot can trim against himself (increase stick forces) within the force band determined by the runaway trim limiter.



The test results show that the stick free and stick fixed neutral points are aft of the rear cg limit of 24 percent MAC between the Mach numbers of .4 and .9 as shown in Fig. 6, Appendix I.

MANEUVERING FLIGHT

The maneuvering flight characteristics (Figs. 7 through 15, Appendix I) are satisfactory for all conditions tested. The stick forces are light compared to most Century Series aircraft; however, some of the stick force per g gradients are in excess of the maximum limits of MIL-F-8785(ASG) as indicated in Fig. 7, Appendix I. The stick force gradient was investigated to the limit load factor at .8 Mach (see Fig. 9, Appendix I) at 10,000 feet as recommended in the Category I report. There is a slight, but acceptable, decrease in the force gradient at high load factors at these and other high "q" conditions. There was no pitch-up tendency during stall buffet other than the normal fall off and regaining of load factor with the stick held aft during heavy stall buffet. The maneuver points were determined during

the Category I tests and were found to be aft of the rear cg limits in all cases.

STALL AND BUFFET BOUNDARY

The 1g and accelerated stall characteristics of the T-38 are very good. A high rate of sink develops rather than a clean nose down pitch during 1g stalls. The buffet boundary preceding the high sink rate at stall is too wide to use as a stall warning.

In the cruise configuration at 1g and 10,000 feet altitude with a gross weight of 10,000 pounds and idle power, initial buffet begins at 190 knots IAS, moderate buffet at 160 knots, heavy buffet and a slow sink rate at 140 knots, and a high sink rate and minimum airspeed occur at 115 knots. A moderate wing roll off occurs at 135 knots and lateral oscillations occur at 125 knots just prior to the development of the high sink rate with full back stick. The wing roll and lateral oscillations provide a satisfactory stall warning.

In the landing configuration at 8800 pounds gross weight, initial buffet begins at 170 knots IAS, moderate buffet at 140 knots, heavy buffet at 130 knots, and a high sink rate and minimum airspeed occur at 112 knots. A moderate wing roll-off occurs at 125 knots and lateral oscillations occur at 120 knots just prior to the development of the high sink rate with full back stick.

Accelerated stall characteristics are similar but occur at higher speeds, dependent upon the load factor present. Recovery from a stall is made by lowering the nose and increasing the airspeed. Stalls and recoveries at 1g and during turning flight did not indicate pitch-up, spin or post stall gyration tendencies. Stall speeds and buffet boundary are presented in Figs. 58 and 59, Appendix I. (C 5)

TABLE I

		Initial Trim Conditions						
Item No.	Altitude ft	Speed V_{cal}-kt	Gear	Flaps deg	Power Setting	Configuration Change	Parameter Held Constant	Maximum Control Forces Experienced lb
1	10,160	212	Down	Up	PLF	Flaps down	Altitude	-6.9
2	25,350	403	Up	Up	MRP	Idle power	Altitude	+3.5
3	25,240	405	Up	Up	MRP	Extend speed brakes	Point of aim	-9.2
4	25,170	331	Up	Up	PLF	Extend speed brakes	Altitude	-8.6
5	25,440	401	Up	Up	MRP	Augmented power	Altitude	+3.2
6	45,170	261	Up	Up	MRP	Idle power	Altitude	+2.5
7	44,910	271	Up	Up	MRP	Extend speed brakes	Point of aim	-3.7
8	45,130	249	Up	Up	MRP	Augmented power	Altitude	+2.5

NOTE: + pull force
- push force

LONGITUDINAL TRIM CHANGES

Tests were conducted to determine the longitudinal trim change resulting from a change in airplane configuration or power setting. The trim changes are easily controllable and all of the forces are less than the 10 pound maximum specified in MIL-F-8785(ASG). The installation of the series trim system, which automatically positions the stabilator as a function of flap position, corrected the objectionable trim change that existed on the prototype aircraft when the flaps were extended.

Safety switches in the horizontal tail system prevent adverse runaway trim by breaking the drive motor circuit in the direction of increasing stick force. The switches were designed to break the circuit at a force of 8 pounds push or pull; however, a force of 3 pounds pull and 9 pounds push were required to break the circuit on the test aircraft. The stick forces required to overcome the effects of runaway trim actuator are shown in Fig. 75, Appendix I. This type of malfunction can only occur if the safety switches mentioned above become welded closed.

LONGITUDINAL SHORT PERIOD STABILITY

The dynamic longitudinal stability of the aircraft is satisfactory with the stability augmenter operative or inoperative. The damping with the augmenter inoperative is not in compliance with MIL-F-8785(ASG) specifications at all airspeeds below an altitude of 30,000 feet; however, damping is considered adequate for mission accomplishment. The damping with the pitch damper operative was unsatisfactory on the prototype aircraft at low airspeed because pitch oscillations were overdamped (some damping ratios were greater than 1.0). The pitch damper was modified and, although the damping is still high at some airspeeds, the damping is acceptable. The longitudinal damping is acceptable. The longitudinal damping characteristics of the T-38 aircraft are non-linear. The damping ratios presented on Figs. 18 and 19, Appendix I, were reduced by a method developed in Reference 6. The damping ratios determined by this method do not show a big increase in damping with the stability augmentation operative. Qualitatively, the pilot feels a marked increase in damping with the augmentation operative.

Pilot inducted oscillations (PIO) are possible in the T-38 aircraft, especially with dampers inoperative at high indicated airspeeds; however, the aircraft can be flown safely without augmentation. Pilot overcontrol can occur at low speeds, especially during landings, due to delayed response of the stabilator and the resulting over-demand of corrective control input. The slow oscillations that result from repeated over-demand of fore and aft control inputs, must be stopped by a deliberate reduction of the control inputs. The high frequency PIO at high airspeeds and the low frequency PIO at low airspeeds demonstrate the compromise necessary to accept a simple control system without a "q" compensator or other "black box" device. There is no failure warning system for an inoperative augmentation condition. An augmentation status and failure warning system and light are required for the rear cockpit (instructor). Without knowledge of augmentation status, an instructor pilot could allow a student to generate overstress conditions (due to low longitudinal damping with augmentation off and possible pilot induced oscillations) before corrective action could be taken. The PIO tendency and its correction should be discussed in the Flight Manual. (C 1)

STATIC DIRECTIONAL STABILITY

The static directional stability characteristics of the T-38 are satisfactory. The aircraft exhibited positive directional stability and positive dihedral effect within the speed range tested. The limited 6 degree rudder travel available in the cruise configuration is adequate for all gear-up directional requirements of the aircraft.

During the static directional tests a roll divergence was encountered in the power approach configuration at low airspeeds (150 knots IAS or less). During sideslips with the gear and flaps down lateral roll oscillations develop with about 20 degrees of rudder (30 degrees available) which gradually diverge and result in an uncontrollable roll in the direction of the applied rudder (see Fig. 33, Appendix I). This characteristic is not considered a problem because it should never be encountered in normal flight, except possibly during maximum crosswind landings where the compensating bank angle accompanied with the large rudder deflections will probably be excessive. A summary of the sideslip characteristics is presented in Fig. 25, Appendix I. (C 6)

It was recommended in the Category I report that the rudder travel be reduced from ±30 to ±20 degrees, if there were no requirements for the larger travel. The contractor reported that in order to reduce the rudder surface travel and still maintain rudder pedal travel within specification requirements, a considerable redesign of the rudder control system would be required. Although there is no flight requirement for the larger rudder travel (except possibly crosswind landings), the technical benefits of reducing the rudder travel are not adequate to justify the cost of the required redesign.

LATERAL-DIRECTIONAL SHORT PERIOD STABILITY

The lateral-directional short period stability characteristics (Dutch roll mode) of the T-38 are satisfactory in all configurations tested, both with and without the yaw damper operative. A slight snaking motion is sometimes encountered at supersonic speeds. A summary of the lateral-directional damping is presented in Fig. 21, Appendix I.

ASYMMETRIC POWER

The asymmetric power characteristics of the T-38 aircraft are very good. Tests conducted at an altitude of 10,000 feet revealed that, with one engine at augmented power and other at idle power, the aircraft can be kept in coordinated flight down to 220 knots IAS with rudder trim only, and to stall with less than maximum rudder deflection without banking the aircraft away from the inoperative engine. One engine throttle chops conducted at the maximum speed (1.1 Mach) and 36,000 feet produced very small directional oscillations and revealed no unsafe conditions.

ASYMMETRIC FLAP

Due to the possibility of an asymmetric flap condition existing on the T-38 aircraft from the failure of a flap motor and the left and right flap interconnect, an investigation of the handling characteristics in this condition was conducted by the contractor and the AFFTC. The contractor's test results are published in Norair report NOR 60-356, dated November 1960. The following AFFTC tests were conducted with the landing gear down at an altitude of 10,000 feet.

TESTS CONDUCTED WITH GEAR DOWN AT 10,000 FEET

Item No.	Indicated Airspeed kt	Flap Position deg	Test Conducted (With Results in Parenthesis)
1	180	LH 45 RH 0	Slowed to minimum control speed (143 knots IAS) checking roll control and rudder required and determining which control was limiting (aileron).
2	180	LH 0 RH 45	Repeated Item 1 (148 knots IAS, aileron).
3	180	LH 45 RH 0	Made final turn maneuvers left and right determining amount of aileron required (32 degrees total left aileron for right turn).
4	160	LH 45 RH 0	Repeated Item 3 (59 degrees total left aileron for left turn, 40 degrees total left aileron for right turn, stick held back under left leg to accomplish left turn).
5	160	LH 20 RH 20	Raise gear and left flap, checking transients and control required to hold wings level as left flap comes up (very little control corrections).
6	160	LH 45 RH 45	Repeat Item 5 (28 degrees total right aileron very little rudder).
7	160	LH 45 RH 45	Right engine at idle, left engine in afterburner. Repeat Item 5 (23 degrees total right aileron required).
8	160	LH 45 RH 45	Repeat Item 7 but raise right flap(full [gear up] aileron, 36 degrees, required until airspeed increases to 170 knots IAS. Roll off is most critical when the flap on the same side as the operative engine stays down.)

Take-offs, landings and go-arounds were performed as follows:

Item No.	Left Flap deg	Right Flap deg	Final Turn Speed, KIAS	Final Speed, KIAS	Touchdown Speed, KIAS	Left Engine Power Setting	Right Engine Power Setting	Speed KIAS	Flap Change
9	0	30	200	180	160				
10	20	20				Afterburner	Afterburner	170	Raise left hand flap
11	0	45	180	170	160				
12	20	20				Military	Military	160	Raise left hand flap
13	0	45	170	160	150				
14	20	20				Afterburner	80 percent	170	Raise left hand flap
15	45	45	170	160	Go-around	Idle	Afterburner	170	Raise left hand flap
16	0	45	170	160	146				

(1) Items 9, 11, 13, 16 are landings, Items 10, 12, 14 are take-offs, Item 15 is a go-around.
(2) The landing gear was raised first on the take-offs except for Item 15.

With less than 600 pounds of fuel aboard, flying in the pattern and landing with one flap full down and the other full up can be accomplished with indicated airspeeds of 170, 160 and 150 knots in the final turn, final approach, and touchdown, respectively. With greater aircraft gross weight, or if turbulent conditions exist on the final approach, the speeds must be increased. Moderate rudder (3 degrees) and full aileron (pilot's leg raised to obtain full stick deflection) were required during a flare and touchdown at 150 knots IAS in jet wash. The forces required are not excessive; however, the stick position (back and under the pilot's leg) is uncomfortable, both physically and psychologically, especially when the aileron stop is met during control movements in rough air. Airspeeds of 180, 170 and 160 knots for turn, final and touchdown, respectively, are more comfortable and safer. If it is possible to raise the down flap, an asymmetric flap approach should be converted to a no flap approach and landing since a no flap approach and landing at the same speeds is safer and more comfortable. (C 8, 9)

Failure of one flap to come up following take-off does not present a serious problem. Roll-off with 20 degrees of flap on one side is mild and decreases rapidly as the aircraft accelerates. The maximum total aileron required during the three take-off tests was 9 degrees. (C 10)

Using the results of test Items 7 and 8 in the above table, go-around and asymmetric flap conditions were set up with the worst engine-flap combination (flap down on side of operative engine). Moderate aileron (16 degrees) was required to combat roll-off but the roll-off decreased as the airspeed increased (see Fig. 66, Appendix I). There was a marked reduction in roll-off with the small increase in speed from 160 knots IAS (as in test Item 8 above) to 170 knots used in the go-around. This single engine asymmetric flap go-around situation could present a serious problem under extreme conditions of low altitude and low airspeed. (C 11)

LATERAL CONTROL

The lateral control characteristics of the T-38 are adequate throughout the entire altitude speed envelope tested. The roll acceleration and roll rate characteristics are excellent (see Fig. 34, Appendix I). The aircraft meets the MIL-F-8785(ASG) roll acceleration requirements of attaining a 90 degree bank angle change from the minimum subsonic combat speed (defined as the best climb speed) to the maximum level flight speed at altitudes tested except above 1.05 Mach at 30,000 feet (there is no roll acceleration requirement above 40,000 feet). The lateral response for formation flight is oversensitive but acceptable.

The aileron-rudder interconnect system (which is explained in detail in Appendix II, page) is designed to engage at an impact pressure of 500 psf, or approximately 370 knots IAS. The interconnect mechanically connects the rudder to aileron motion such that $\delta_r / \delta_a = .22$ in the direction to "uncoordinate" rolling maneuvers. This system is necessary to prevent structural failures due to the high loads anticipated during some rolling maneuvers. In the event of an electrical failure or solenoid failure, this system is designed to engage so that flight at high speed will be safe and full lateral inputs can be made without fear of structural failures. Because of this fail safe design feature, flight below 370 knots with the aileron-rudder interconnect engaged may be quite common. (Numerous failures of the interconnect solenoid have occurred in the flight test program to date.)

Tests with the interconnect solenoid in the failed position were conducted to investigate the roll characteristics at low speeds and determine if any unsafe conditions existed. The aircraft can be flown safely in all configurations throughout the flight envelope with the interconnect engaged. The roll rate at low airspeeds is lower with the interconnect engaged that with it disengaged. This is due to the adverse yaw caused by the rudder input with the interconnect engaged (see Figs. 55 and 56, Appendix I). The decrease in roll rate with the interconnect engaged is most noticeable while performing rolls initiated from high load factor turns. Landings were made with the interconnect engaged and no adverse effects were noted.

An analog computer investigation of the roll characteristics in the landing configuration was made by AFFTC. This study revealed no unsafe roll characteristics with or without the interconnect engaged. The results were published in AFFTC-TN-60-13.

The effect of the interconnect at high airspeeds is to reduce the amount of complimentary yaw in rolls and thus reduce the vertical tail loads (see Fig. 35, Appendix I). The contractor's estimated data indicated that this condition would be most critical during rolls from high entry load factors at high airspeeds. A roll coupling program is presently being conducted by the contractor to investigate roll coupling characteristics and the effects of the interconnect. (C 7)

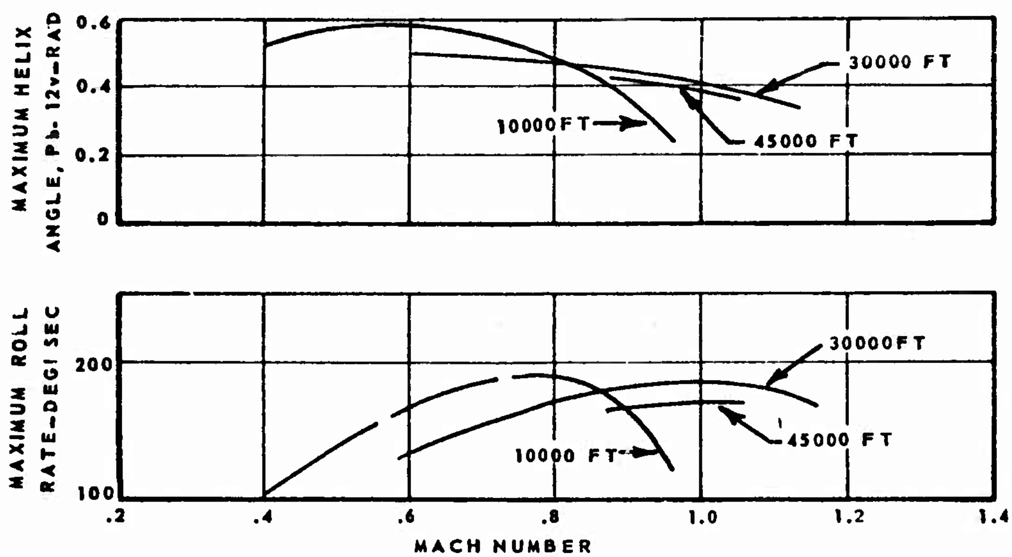
The present aileron control system incorporates a 19.6 pound spring stop to limit the gear-up total aileron travel to 32.5 degrees. The spring stop can be overpowered so that a total of 60 degrees of aileron travel can be obtained. The spin tests on the aircraft have demonstrated that spin recovery is most positive with as much anti-spin aileron (ailerons with the spin) as can be obtained. The contractor is conducting tests to determine if the spring stop can be removed throughout the entire flight envelope.

SPIN CHARACTERISTICS

The T-38 aircraft will be prohibited from spins in service use. The spin characteristics were investigated on an aircraft equipped with a spin recovery chute to determine if recovery from unintentional spins is possible.

The spin characteristics of the T-38 will be reported in detail in a separate report. The aircraft will spin from

AILERON ROLL SUMMARY



both erect and inverted entries, and has both an inverted and erect spin mode. The erect spin may be either oscillatory or smooth and flat. Recovery from well developed oscillatory spins is slow and can not be assured without the installation of an emergency hydraulic system (both engines may flameout resulting in complete loss of hydraulic pressure and thus control power). The aerodynamic controls are not effective in flat spins and recovery from this type spin is not assured.

APPROACH AND LANDING

The handling characteristics of the T-38 in the landing pattern are satisfactory and provide adequate Century Series aircraft simulation. Although the base leg, final turn, and final approach are flown in light to moderate stall warning buffet, the wing roll and lateral instability at speeds closer to the stall provide adequate stall warning both in 1g and turning flight. The slow response to longitudinal stick inputs can generate a low frequency oscillation which can be halted through use of smaller control inputs.

The nose can be held off after touchdown up to speeds near 110 knots IAS. Touch and go landings can be accomplished with full flaps left down, landing trim, and the nose wheel held off throughout the ground roll.

CONTROL BREAK-OUT AND FRICTION FORCES

The control break-out and friction forces are all satisfactory, although the directional break-out force is in excess of MIL-F-8785(ASG) limits. Control system characteristics are presented in Figs. 69 through 73, Appendix I. A slight unsymmetric directional break-out force existed on the test aircraft, but was not objectionable. The friction band was noticeable only during level flight acceleration or deceleration where the small stick force changes are masked in the friction band. The aileron spring detent for the gear-up aileron deflection limit was changed from the production value of 19.6 pounds to an experimental value of 9 pounds. This was done to determine if the production spring stop could be reduced and still provide a positive stop. The lighter spring proved

to be not strong enough to provide a positive stop and was eliminated from consideration for production use. The aileron force characteristics presented in Fig. 71, Appendix I, were obtained with the 9 pound spring installed. Production aileron force characteristics with the 19.6 pound spring are presented in Fig. 88, Appendix I, AFFTC-TR-59-42).

AIRSPEED CALIBRATION

The test nose boom system was calibrated by the tower fly-by, pacer, and smoke trail acceleration methods (see Fig. 74, Appendix I). The production airspeed system has been modified as recommended in the Category I report, AFFTC-TR-59-42, and will include a short nose boom. Development testing of the production airspeed system is presently being conducted by the contractor. The final production airspeed system will be calibrated on the Category II performance test aircraft.

CONCLUSIONS

The T-38 aircraft is an excellent trainer for pilot transition into high performance jet aircraft including present Century Series fighter aircraft. Its simplicity is conducive to transition; its performance and handling characteristics allow Century Series simulation and development of correct pilot techniques. Visibility from the front seat and the rear seat is excellent during all phases of flight. The rear seat (instructor's position) affords complete control over most of the normal and emergency procedures.

The stability and control characteristics of the aircraft are satisfactory throughout the flight envelope. Most of the undesirable features noted during the Category I tests have been improved or corrected. The longitudinal control

response is sensitive at high speeds and/ or Mach numbers and slow at airspeeds below 220 knots IAS, but it is the best balance available from the simple non-q-biased flight control system. The aircraft is safe to fly with or without stability augmentation (pitch and yaw dampers), although the damping of longitudinal disturbances with the pitch damper inoperative is low and military specification requirements are not met in almost all flight conditions. The damper systems provide adequate damping at all airspeeds and slightly excessive damping longitudinally at low airspeeds. All in-flight directional lateral-directional, and lateral characteristics are satisfactory. The aircraft has excellent roll acceleration and roll rate characteristics. There is a wide pre-stall buffet boundary which starts at airspeeds as high as 190 knots IAS in the cruise configuration and 170 knots in the power approach configuration, and continues to the minimum airspeed of about 115 knots. The buffet is unsatisfactory as a stall warning; however, lateral roll-off near the stall and lateral oscillations which occur just prior to stall provide a satisfactory stall warning.

An asymmetric flap condition can exist in the T-38 aircraft from the failure of a flap motor and the left and right flap interconnect. The aircraft can be landed with one flap full down and the other flap up. The safe landing pattern airspeeds under this condition are almost the same as those airspeeds for a flaps up landing. A flaps up pattern is safer and easier to fly and should be selected if the asymmetric flap condition can be converted to a flaps up condition. An asymmetric flap condition following a take-off with a take-off flap setting of 20 degrees does not present a problem. The wings can be held level with a moderate amount of aileron at airspeeds as low as 160 knots IAS. An asymmetric condition during a go-around does not present a serious control problem unless it is combined with a single engine condition with the operative engine on the same side as the down flap (even under these conditions full aileron is not required to maintain a wings level condition at airspeeds above 170 knots).

The low nose wheel steering response near neutral requires excessive rudder pedal movements while taxiing and occasionally causes a low frequency divergence in directional oscillations on take-off due to overcontrol. This divergence can be stopped by releasing the nose wheel steering. There are a number of discrepancies in the cockpit layout and controls which should be corrected. None of these are unacceptable if adequately discussed in the Pilot's Handbook.

RECOMMENDATIONS

A. The following recommendations are for correction of safety of flight items and should be accomplished before the aircraft goes into operational use[3]:

*1. Provide a positive throttle stop at idle to prevent inadvertent fuel cut-off (page 4

**2. Reduce the intensity of the land-gear position lights, the J-4 compass light and the fuel boost pump and crossfeed lights at the dimmed setting for night operation (page 4

*3. Adjust the fuel remaining gages to show fuel actually available for engine use. The engines should not flame-out until a reading of zero is seen (page 4

**4. Reduce the reflection problem in the rear cockpit caused by the transparent blast shield between the cockpits (page 4

*5. Change the emergency landing gear extension mechanism to provide a decrease in forces near the end of the cable travel and a single pull for actuation (page 4

6. Include a standby magnetic compass in the rear cockpit (page 4

*7. Equalize the interphone volume of feedback to the speaker and reception to the listener (page 4

[3]* Indicates that corrective action has been initiated or approved.

** Indicates that the item is under consideration by T-38 Project Office.

B. The following recommendations are for correction of undesirable deficiencies and/or improvement of the aircraft:

*1. Increase the size and improve the detent action of the speed brake control switch in the front cockpit (page 4

*2. Relocate the stability augmentation switches to behind the throttle quadrant (page 4

*3. Provide an indication in the rear cockpit of damper system status (page 7

**4. Provide an armrest on the left side of the cockpit (page 7

*5. Reduce the throttle sweep angle by at least one third, using the middle range of the present sweep (page 7

6. Increase the seat response to switch actuation (page 7

7. Increase the extension and retraction rates of the landing light(*) and the speed brake (page 7

*8. Provide for an emergency engine fuel cut-off system that is separate from throttle action (page 7

9. Add a canopy handle to the left side of the canopy to provide a hand grip and reduce wear and tear on the defrost ducts (page 7

*10. Improve the air conditioning control system to prevent loud cycling and pressure fluctuation (page 7

*11. Find and correct the cause for the loud "airhorn" type noise which develops during high speed dives (page 7

**12. Tilt the warning light panel against the side of the cockpit to provide better visibility and less obstruction by the pilot's knee (page 7

**13. Permit engine operation in the 50 to 58 percent rpm range for taxiing (page 8

**14. Increase the nose wheel steering response to small rudder deflections about neutral (page 8

**15. Reduce the pedal force required to hold the aircraft at military rated power (page 8

C. The following specific flight conditions should be discussed in the Flight Manual in addition to the normal subjects:

*1. Dampers-off flight is safe but pilot induced oscillations are possible. Correction for pilot induced oscillation is to release controls (page 12

**2. During dusk and night flights reflection from the blast shield to the occupant of the rear seat can cause fake bogies, disorientation and vertigo (page 4

*3. Loud airhorn noises may be generated during high speed dives (page 7

**4. Nose wheel steering system characteristics can cause divergent directional oscillations during take-off. The correction is to turn-off nose wheel steering (page 8

*5. The approach to stall is indicated by a wide buffet margin, a wing drop tendency nearer the stall, and a lateral oscillation just prior to the stall (page 11

*6. A roll-off in the direction of applied rudder pedal may be encountered during low speed sideslips (page 13

*7. The aircraft can be flown and landed with the aileron-rudder interconnect failed (engaged during low speed flight) (page 16

*8. If possible, an asymmetric flap condition should be converted to a no flap condition for landing (page 15

*9. If an asymmetric flap landing must be made, the recommended minimum speeds are 180 knots IAS in the final turn, 170 knots on final, and 160 knots at touchdown. Up to full aileron may be required (page 15

*10. An asymmetric flap condition after take-off, occurring as take-off flaps are raised, does not present a problem (page 15

*11. An asymmetric flap condition during a go-around, occurring as full flaps are raised, does not present a problem unless combined with a single engine condition with the operative engine on the same side as the down flap. For this condition, full aileron is required at speeds up to 170 knots IAS (page 15

APPENDIX I

data analysis methods

REFERENCES

The data presented in this report was reduced and analyzed using the following references:

1. MIL-F-8785(ASG), "Flying Qualities of Piloted Aircraft."

2. AFFTC-TR-59-21, "AFFTC Stability and Control Techniques."

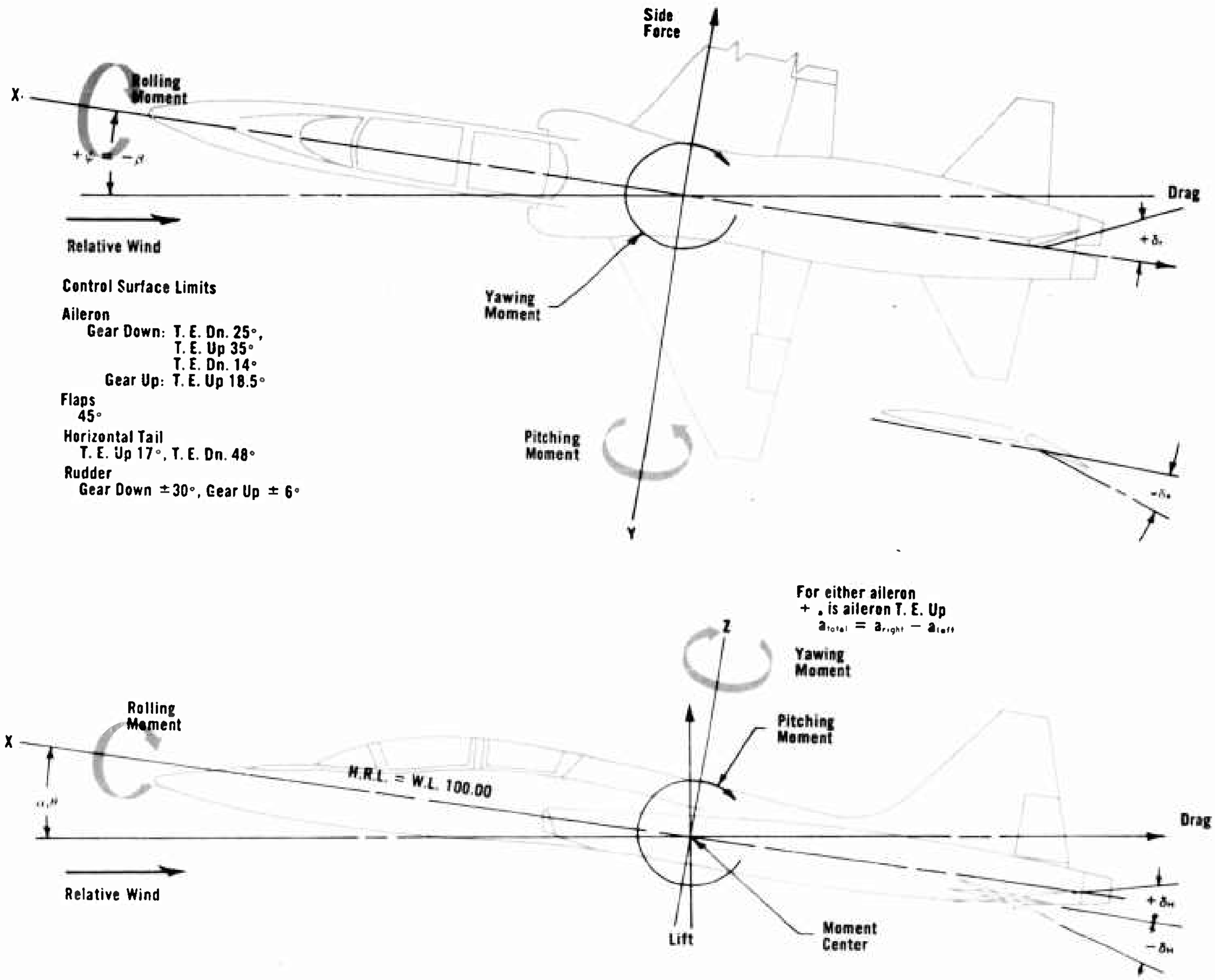
3. U.S. Standard Atmosphere.

4. AFFTC-TR-59-42, "Category I Flight Test."

5. AFFTC-TN-60-13, "Analog Computer Study of the T-38 Airplane in the Landing Configuration."

*6. Norair Memorandum, "Evaluation of Airplane Transients," dated 4 September 1959.

*NOTE: The T-38 aircraft has non-linear damping characteristics which complicates the obtaining of damping ratio data from dynamic pulses. This reference was used as the aid in deducing the damping characteristics presented in this report. The damping ratios reduced by the method outlined in this reference do not show the definite increase in damping noted by the pilot when the dampers are operative.



Side Force
X
Rolling Moment
+ψ −β
Drag
$+\delta_r$
Relative Wind
Control Surface Limits
Aileron
Gear Down: T. E. Dn. 25°,
T. E. Up 35°
T. E. Dn. 14°
Gear Up: T. E. Up 18.5°
Flaps
45°
Horizontal Tail
T. E. Up 17°, T. E. Dn. 48°
Rudder
Gear Down ±30°, Gear Up ± 6°
Yawing Moment
Pitching Moment
Y
$-\delta_a$
For either aileron
+ , is aileron T. E. Up
$a_{total} = a_{right} - a_{left}$
Z
Yawing Moment
Rolling Moment
X
Pitching Moment
H.R.L. = W.L. 100.00
α, θ
Drag
Relative Wind
$+\delta_H$
$-\delta_H$
Lift
Moment Center

NOMENCLATURE

ARI	aileron-rudder interconnect	
b	wing span	ft
cg	center of gravity	%MAC
$C_{1/2}$	cycles to damp to one-half amplitude	
$C_{1/10}$	cycles to damp to one-tenth amplitude	
C_N	normal force coefficient, $.000675\,n\,(W/\delta_a)/M^2S$	
F_a	lateral stick force	lbs
F_r	rudder pedal force	lbs
F_s	longitudinal stick force	lbs
g	acceleration due to gravity	ft/sec^2
H_p	pressure altitude	ft
H_{ic}	instrument corrected indicated altitude	ft
M	free stream Mach number	
n	normal load factor	g
n_y	lateral load factor	g
q	dynamic pressure	lb/ft^2
q_c	impact pressure	lbs/ft^2
S	wing area	ft^2
T_a	ambient temperature	°K
V_c	calibrated airspeed	kt
V_e	equivalent airspeed	kt
V_{ic}	instrument corrected indicated airspeed	kt
V_t	true airspeed	kt
W	airplane gross weight	lb
α	angle of attack	deg
β	angle of sideslip	deg
$\delta_{a_{tot}}$	*total aileron deflection $=\delta_{a_R}-\delta_{a_L}$	deg
δ_r	rudder deflection	deg
δ_s	stabilizer deflection	deg
δ_a	$P_a/29.92$	
$\Delta P/q$	pressure coefficient	
ξ	damping ratio	
θ	pitch angle	deg
$\dot{\theta}$	pitch rate	deg/sec
$\ddot{\theta}$	pitch acceleration	deg/sec^2
τ	period	sec
ϕ	bank angle	deg
$\dot{\phi}$	roll rate	deg/sec
$\ddot{\phi}$	roll acceleration	deg/sec^2
$\dot{\psi}$	yaw rate	deg/sec

The subscript i denotes an indicated condition.

*For either aileron $+\delta_a$ is aileron trailing edge up. $+\delta_{a_{tot}}$ is for a right roll. $-\delta_{a_{tot}}$ is for a left roll.

STABILITY AND CONTROL PLOTS

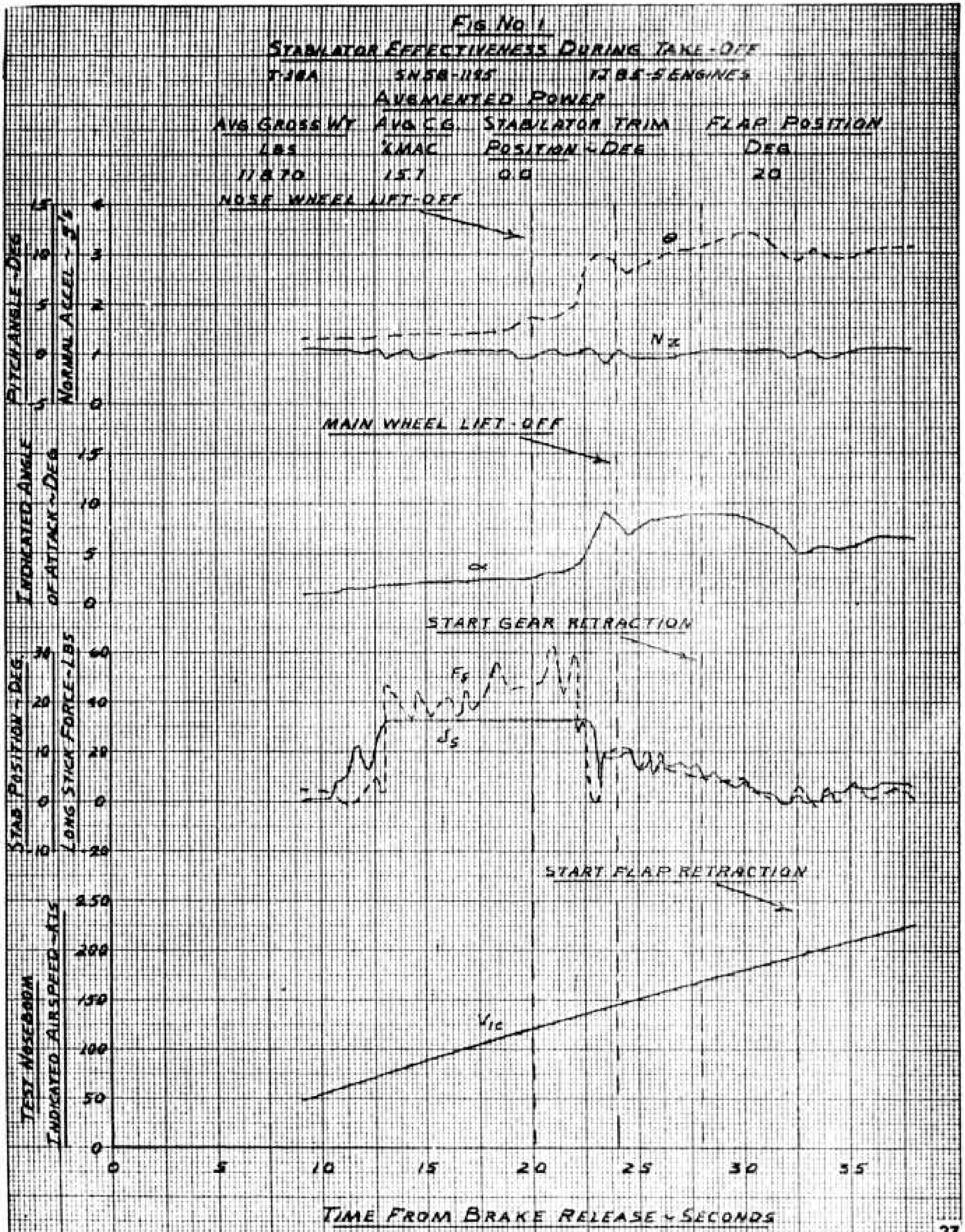


FIG NO 1
STABILATOR EFFECTIVENESS DURING TAKE-OFF
T-38A SN58-1195 J85-5 ENGINES
AUGMENTED POWER
AVG GROSS WT LBS 11870
AVG C.G. %MAC 15.7
STABILATOR TRIM POSITION ~ DEG 0.0
FLAP POSITION DEG 20
NOSE WHEEL LIFT-OFF
MAIN WHEEL LIFT-OFF
START GEAR RETRACTION
START FLAP RETRACTION
PITCH ANGLE - DEG
NORMAL ACCEL - g's
INDICATED ANGLE OF ATTACK - DEG
STAB POSITION - DEG
LONG STICK FORCE - LBS
TEST NOSEBOOM INDICATED AIRSPEED - KTS
θ
N_z
α
F_s
δ_s
V_{ic}
TIME FROM BRAKE RELEASE ~ SECONDS

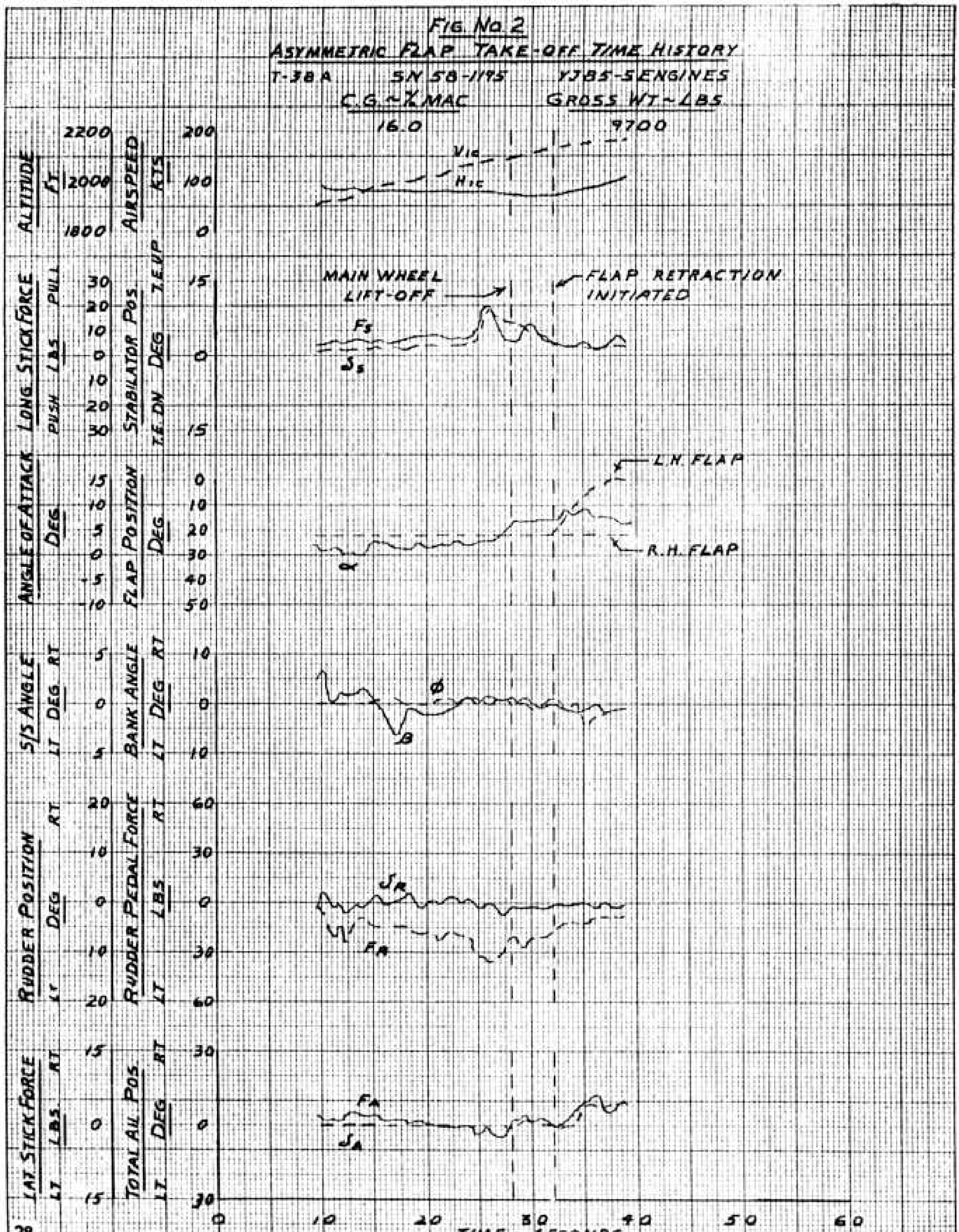


FIG. No. 2
ASYMMETRIC FLAP TAKE-OFF TIME HISTORY
T-38A SN 58-1195 YJ85-5 ENGINES
C.G. ~ % MAC GROSS WT ~ LBS
16.0 9700
ALTITUDE FT
AIRSPEED KTS
LONG. STICK FORCE PUSH LBS PULL
STABILATOR POS. T.E.DN DEG T.E.UP
ANGLE OF ATTACK DEG
FLAP POSITION DEG
S/S ANGLE LT DEG RT
BANK ANGLE LT DEG RT
RUDDER POSITION LT DEG RT
RUDDER PEDAL FORCE LT LBS RT
LAT. STICK FORCE LT LBS RT
TOTAL AIL. POS. LT DEG RT
MAIN WHEEL LIFT-OFF
FLAP RETRACTION INITIATED
L.H. FLAP
R.H. FLAP
TIME ~ SECONDS

FIG. NO. 3

LONGITUDINAL SPEED STABILITY

T-38A SN 58-1195 YJ85-5 ENGINES

CRUISE CONFIGURATION

SYMBOL	MACH NO.	TRIM Vc ~ KTS	Hp ~ FT.	AVG. GROSS WT.	AVG. C.G. ~ % MAC	TRIM δ STAB
○	.923	518	10340	11230	15.0	1.1° T.E. UP
□	.919	517	10190	10340	21.1	.7° T.E. UP

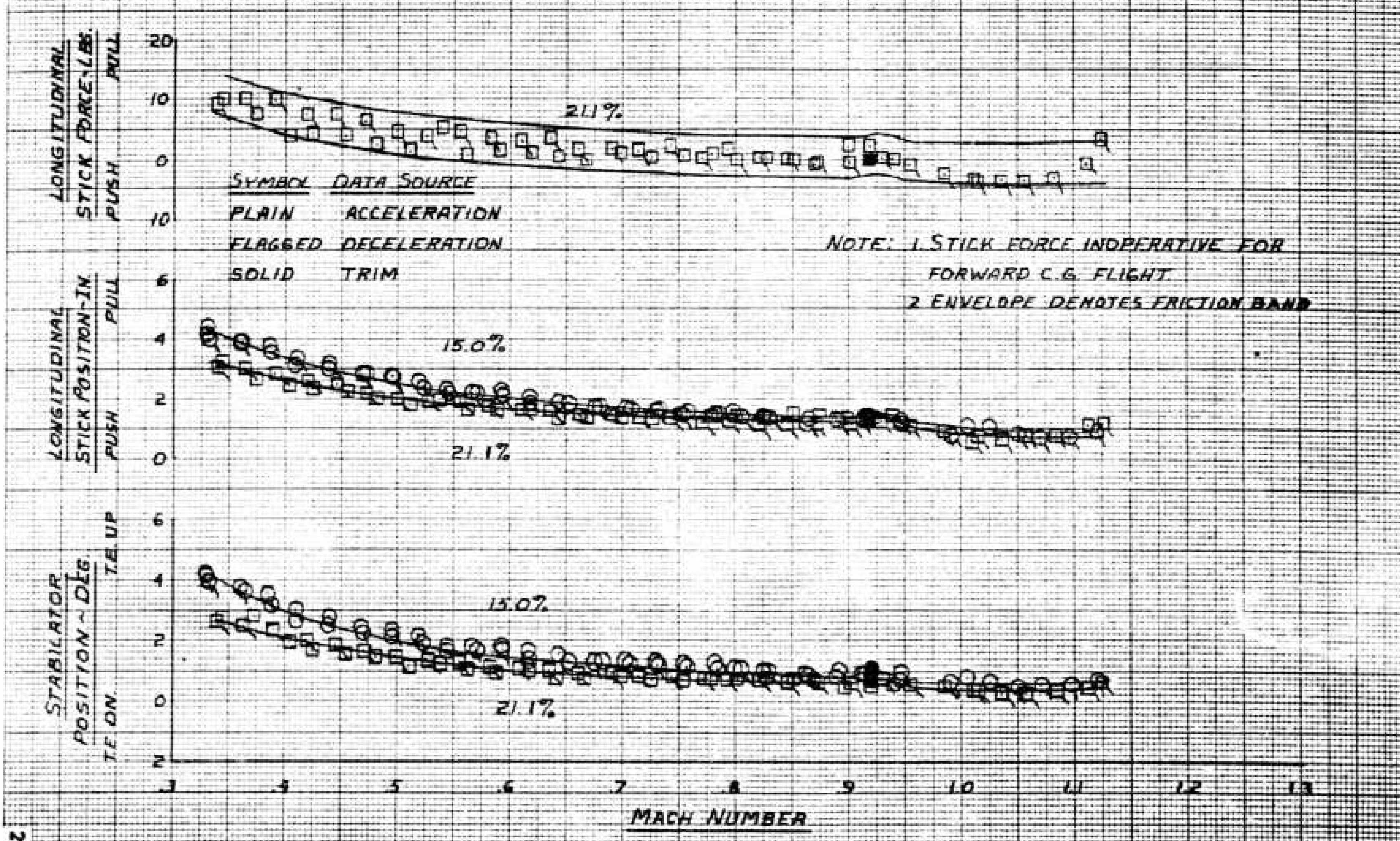


FIG No. 4

LONGITUDINAL SPEED STABILITY

T-38A SN 58-1195 YJ85-5 ENGINES

CRUISE CONFIGURATION

SYMBOL	TRIM Vc KTS.	HP FT	MACH NO.	AVG. GROSS WT. LBS.	AVG. C.G. % MAC.	TRIM STAB
O	472	25430	1.092	9230	16.0	.9° T.E. UP
□	471	25610	1.092	9430	24.3	.5° T.E. UP

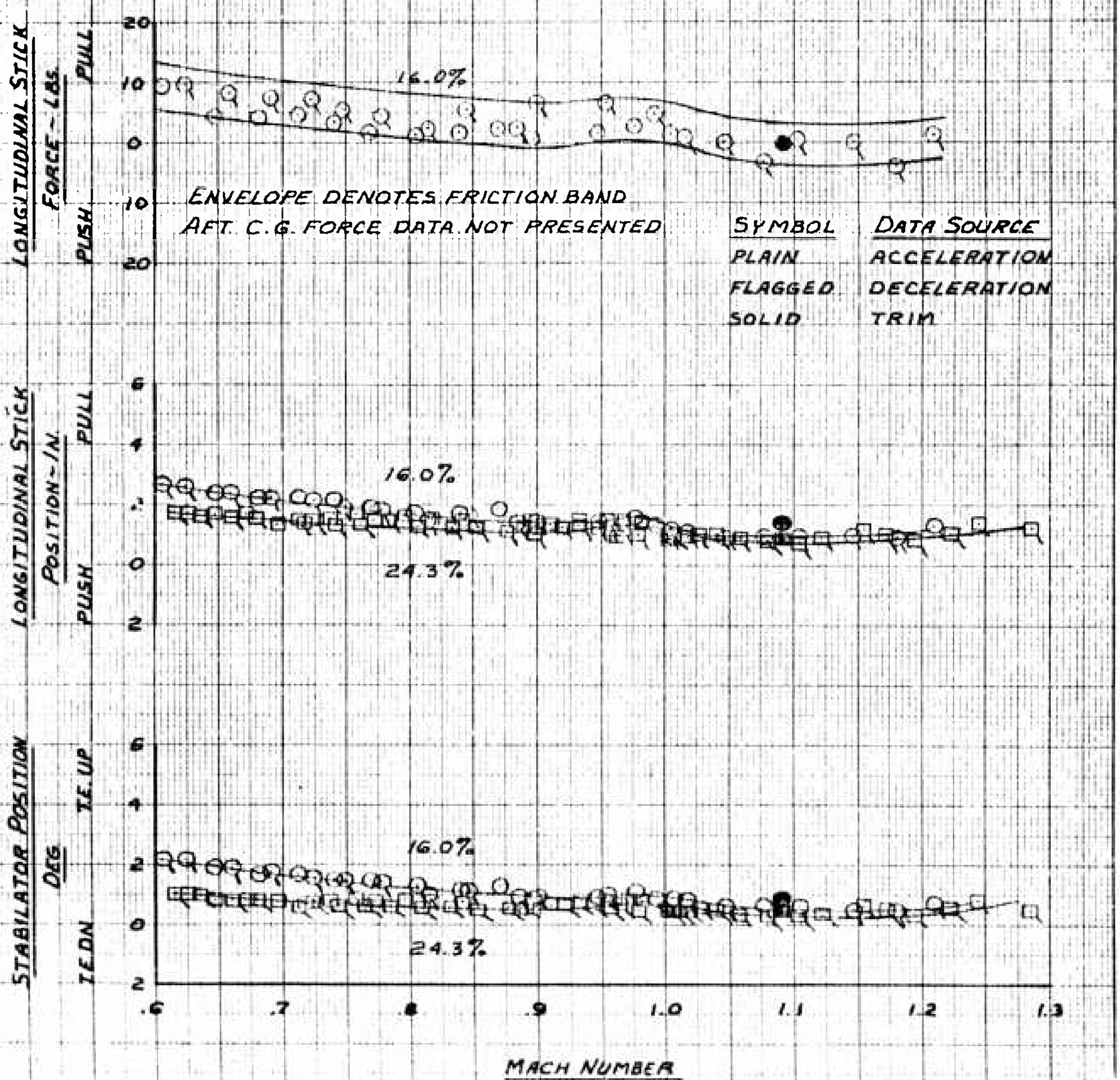


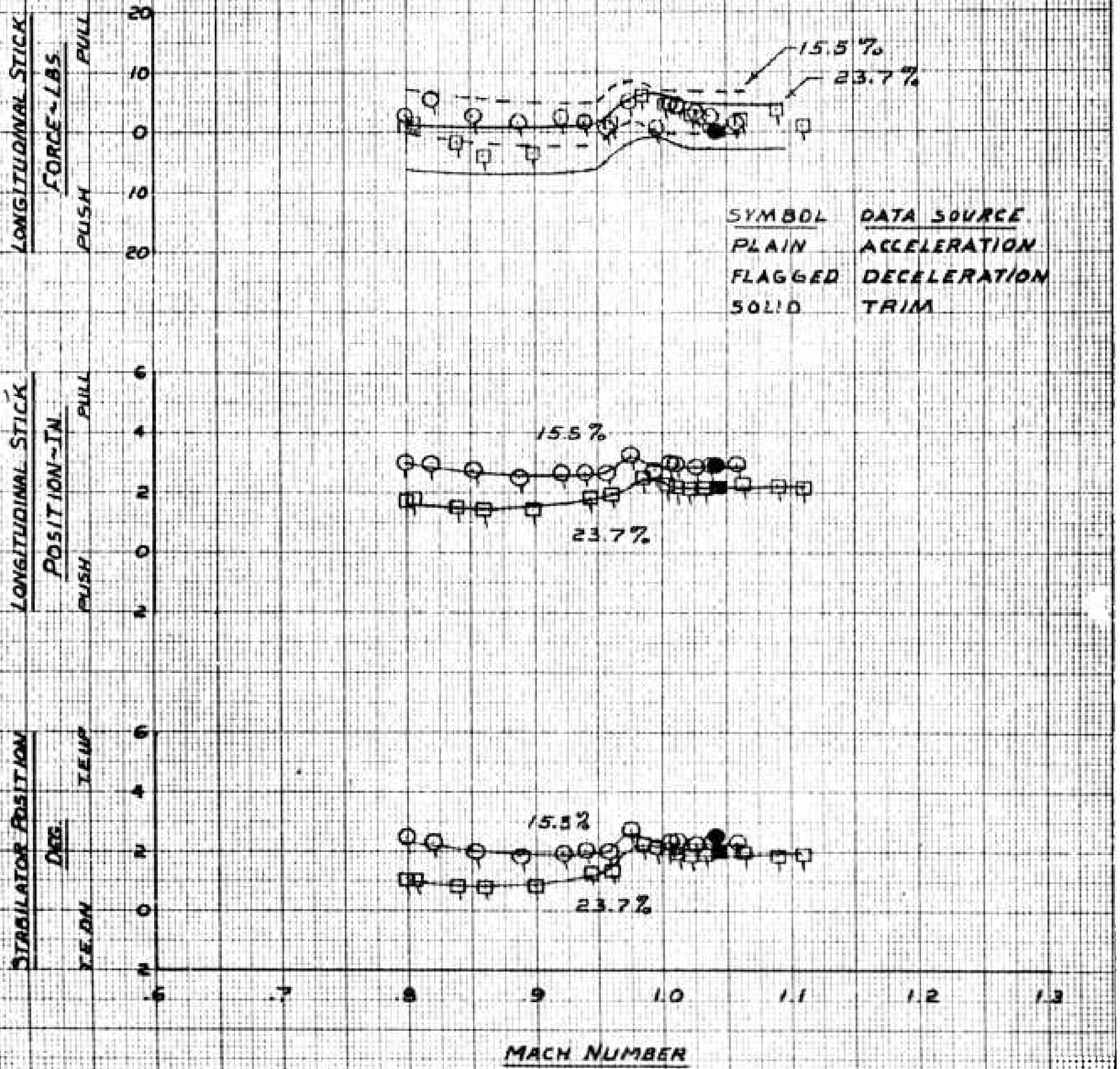
FIG. NO. 5

LONGITUDINAL SPEED STABILITY

T-38A SN 58-1195 YJ85-5 ENGINES

CRUISE CONFIGURATION

SYMBOL	TRIM Vc KTS.	Hp FT	MACH NO.	AVG. GROSS WT LBS.	AVG. C.G. % MAC.	TRIM & STAB
○	292	44870	1.041	9640	15.5	2.8° T.E. UP
□	294	44870	1.043	9300	23.7	2.0° T.E. UP



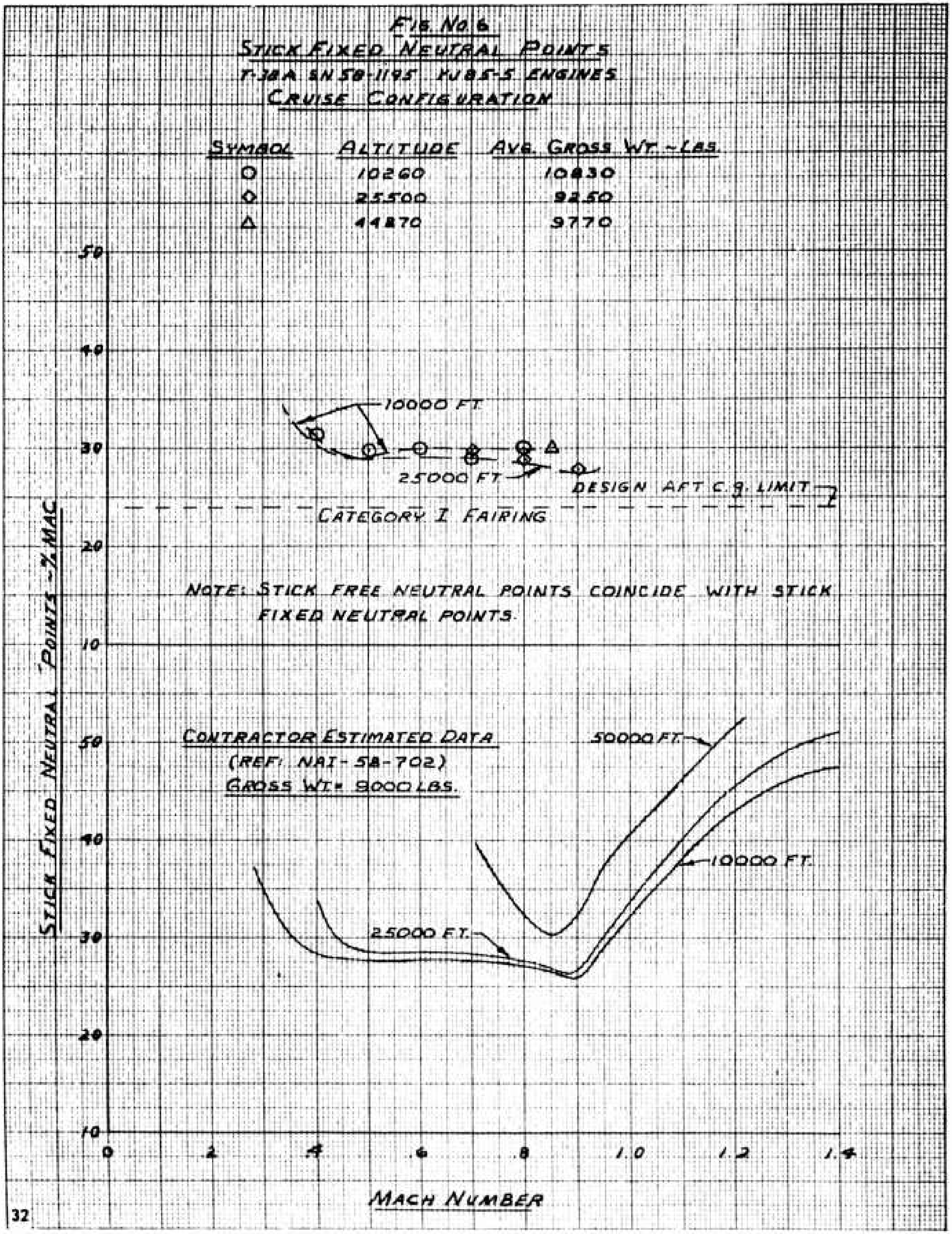


FIG. No. 6
STICK FIXED NEUTRAL POINTS
T-38A SN 58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION
SYMBOL ALTITUDE AVG. GROSS WT ~ LBS
○ 10260 10830
◇ 25500 9250
△ 44870 9770
10000 FT.
25000 FT.
DESIGN AFT C.G. LIMIT
CATEGORY I FAIRING
NOTE: STICK FREE NEUTRAL POINTS COINCIDE WITH STICK FIXED NEUTRAL POINTS.
CONTRACTOR ESTIMATED DATA
(REF: NAI-58-702)
GROSS WT = 9000 LBS.
50000 FT.
10000 FT.
25000 FT.
STICK FIXED NEUTRAL POINTS ~% MAC
MACH NUMBER
50
40
30
20
10
50
40
30
20
10
0
.2
.4
.6
.8
1.0
1.2
1.4

FIG. NO. 7
STICK FREE MANEUVER POINTS
T-38A SN58-1195 YJ85-5 ENGINES

SYMBOL	CONFIGURATION	ALTITUDE~FT	GROSS WT~LBS.
○	CRUISE	10000	8900-11300
□	CRUISE	25000	9800-10300
△	CRUISE	45000	8900-10300

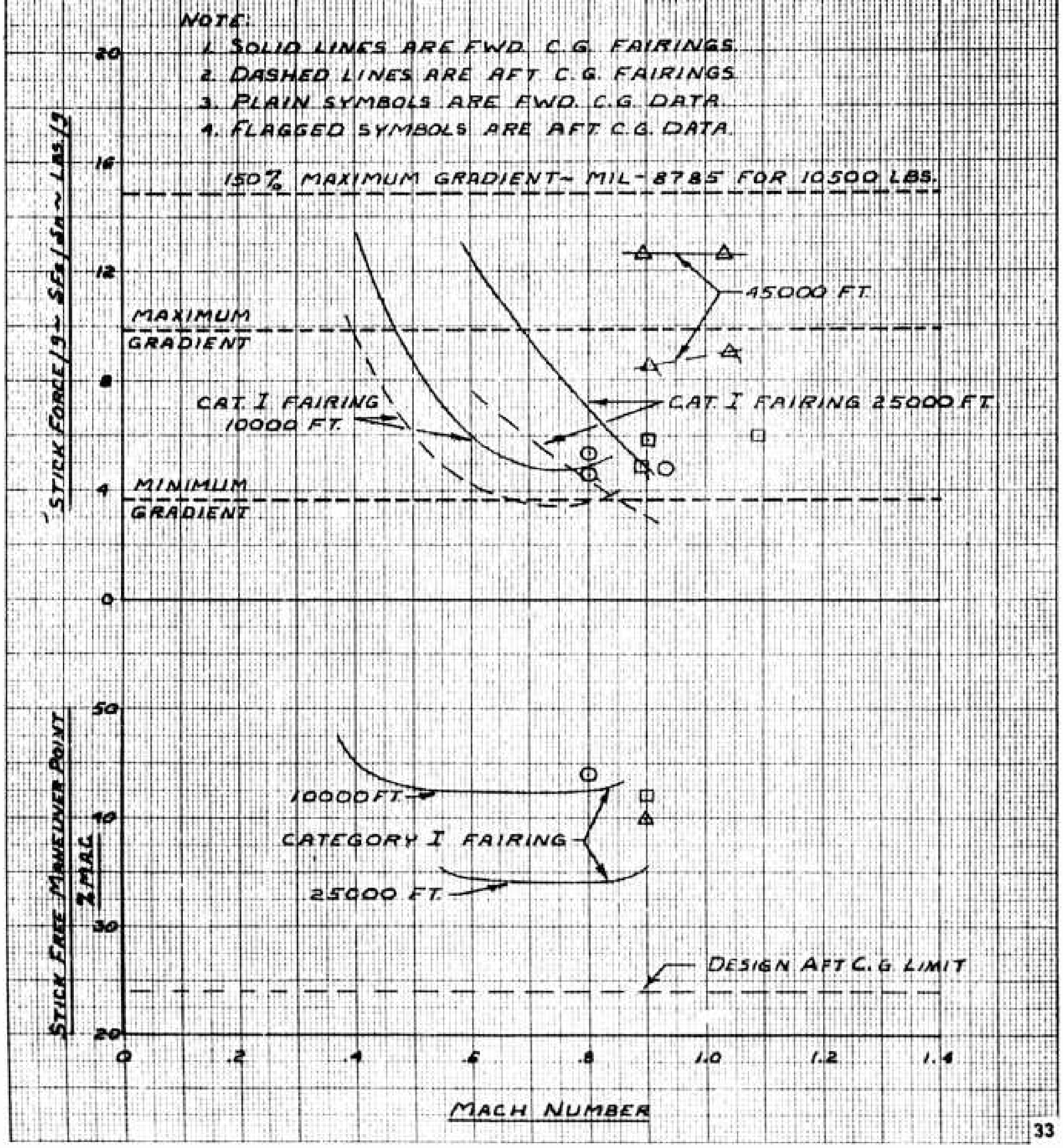
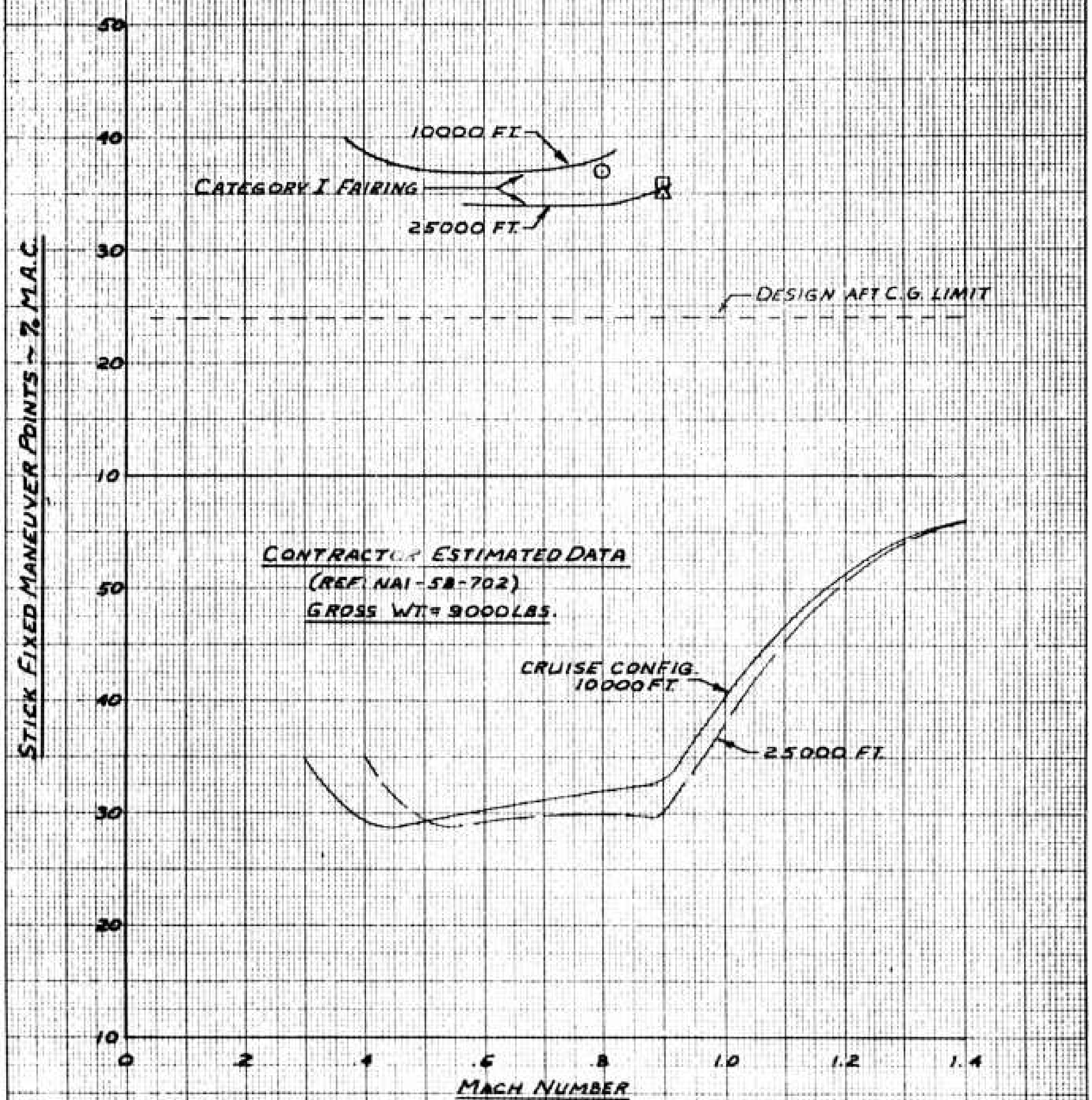


FIG. NO. 8

STICK FIXED MANEUVER POINTS

T-38A SN58-1195 YJ85-5 ENGINES

SYMBOL	CONFIGURATION	ALTITUDE ~ FT.	GROSS WT. ~ LBS.
○	CRUISE	10000	8500-11300
□	CRUISE	25000	9800-10300
△	CRUISE	45000	8900-10300



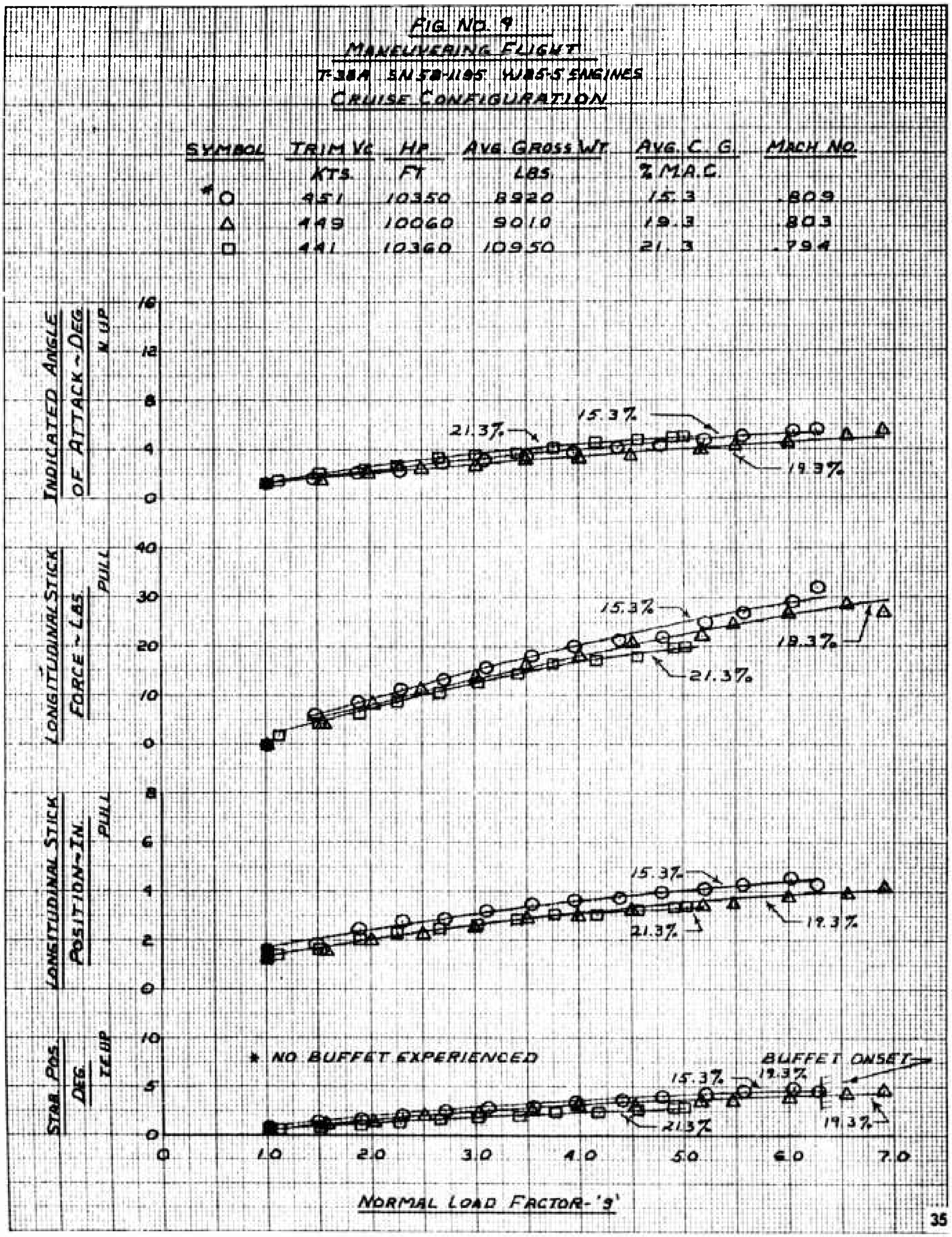


FIG. NO. 9
MANEUVERING FLIGHT
T-38A SN 58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION
SYMBOL
TRIM Vc KTS.
Hp FT
AVG. GROSS WT LBS.
AVG. C.G. % M.A.C.
MACH NO.
451
10350
8920
15.3
.809
449
10060
9010
19.3
.803
441
10360
10950
21.3
.794
INDICATED ANGLE OF ATTACK ~ DEG
N UP
LONGITUDINAL STICK FORCE ~ LBS.
PULL
LONGITUDINAL STICK POSITION ~ IN.
PULL
STAB. POS. DEG.
TEUP
15.3%
19.3%
21.3%
* NO BUFFET EXPERIENCED
BUFFET ONSET
NORMAL LOAD FACTOR - 'g'

FIG. NO. 10

MANEUVERING FLIGHT

T-38A SN58-1195 YJ85-5 ENGINES

CRUISE CONFIGURATION

SYMBOL	TRIM Vc KTS	HP FT	AVG. GROSS WT. LBS.	AVG. C.G. % M.A.C.	MACH NO.
* ○	530	9760	10120	16.8	.935
△	527	10510	9500	19.8	.940
□	527	10160	11310	21.2	.936

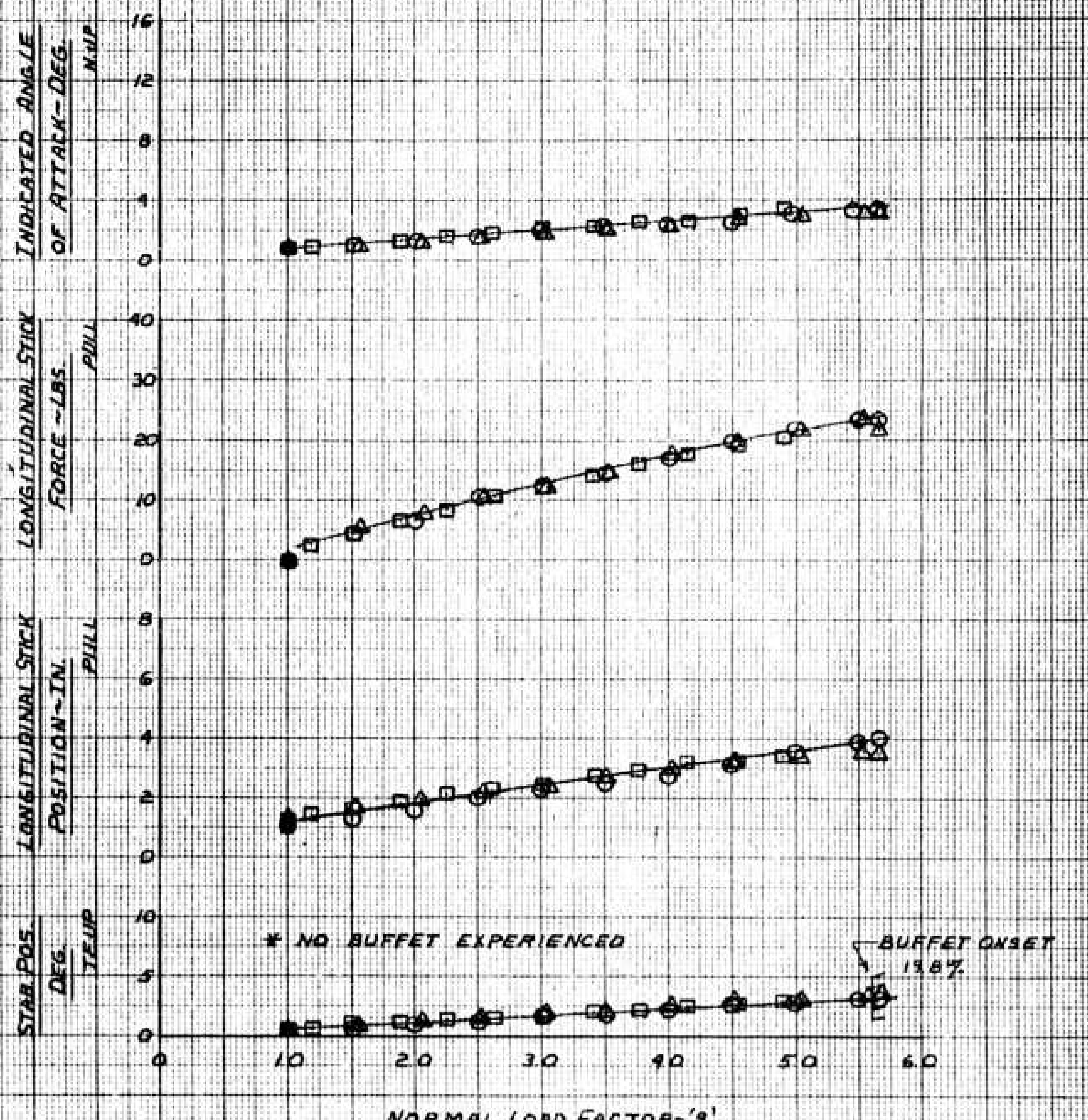


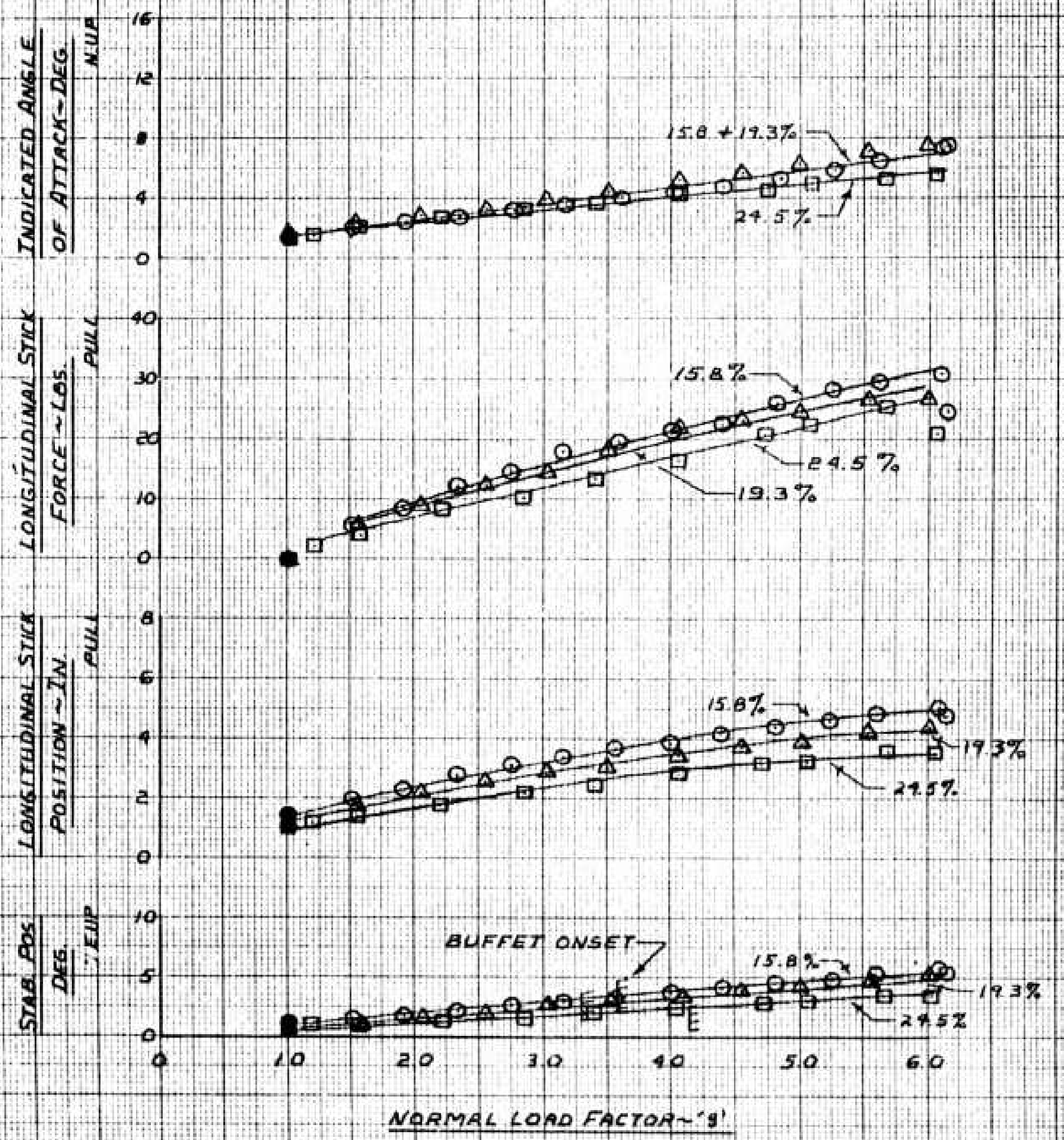
FIG. NO. 11

MANEUVERING FLIGHT

T-38A SN58-1195 YJ85-5 ENGINES

CRUISE CONFIGURATION

SYMBOL	TRIM Vc KTS.	HP FT	AVG. GROSS WT. LBS.	AVG. C.G. % M.A.C.	MACH NO.
○	385	25020	9090	15.8	.903
△	388	25190	9920	19.3	.911
□	381	24780	9040	24.5	.890



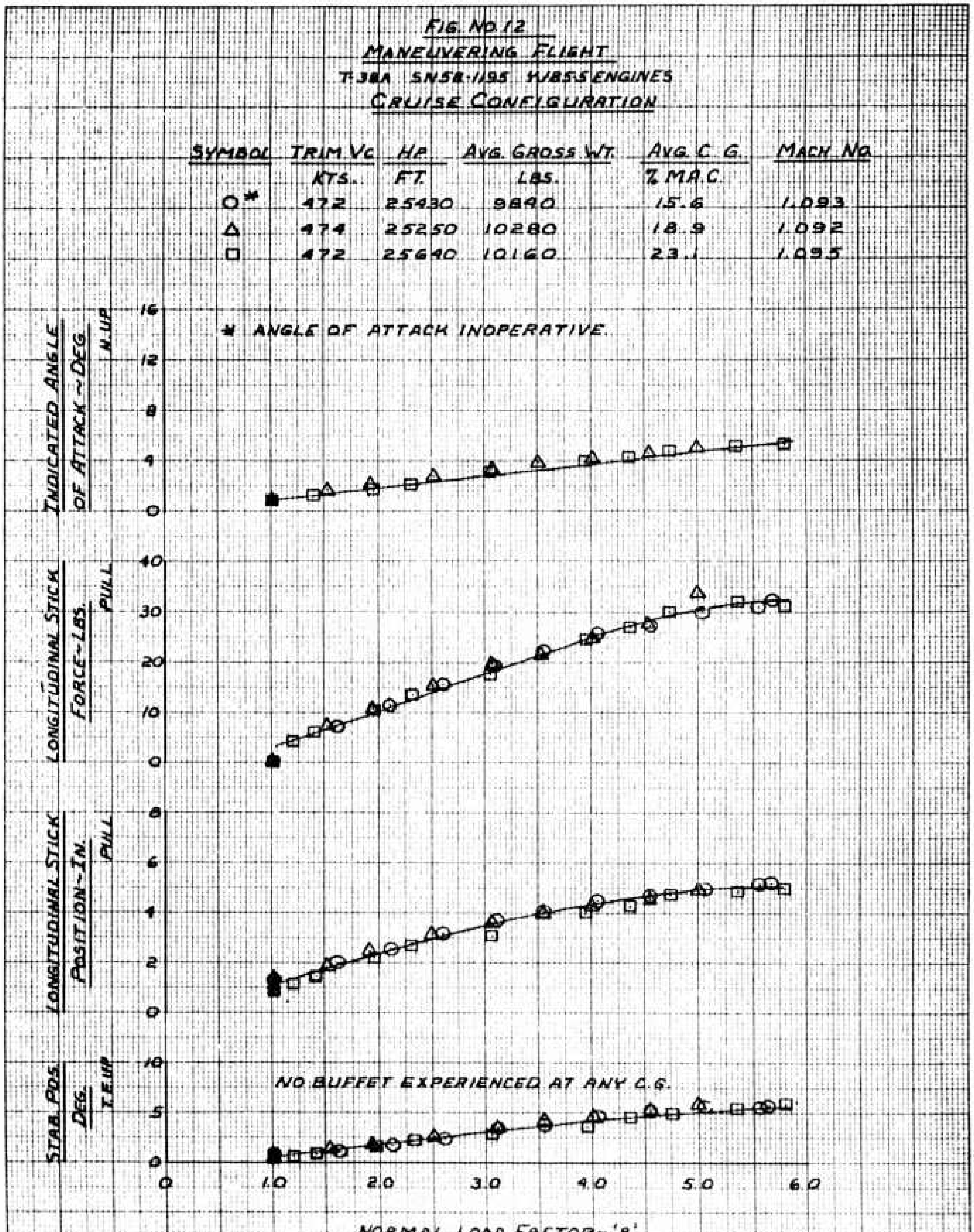


FIG. NO 12
MANEUVERING FLIGHT
T-38A SN58-1195 J85 ENGINES
CRUISE CONFIGURATION
SYMBOL TRIM Vc KTS. HP FT. AVG. GROSS WT. LBS. AVG. C.G. % M.A.C. MACH NO.
○* 472 25430 9840 15.6 1.093
△ 474 25250 10280 18.9 1.092
□ 472 25640 10160 23.1 1.095
* ANGLE OF ATTACK INOPERATIVE.
INDICATED ANGLE OF ATTACK ~ DEG. N.UP
LONGITUDINAL STICK FORCE ~ LBS. PULL
LONGITUDINAL STICK POSITION ~ IN. PULL
STAB. POS. DEG. T.E.UP
NO BUFFET EXPERIENCED AT ANY C.G.
NORMAL LOAD FACTOR ~ 'g'

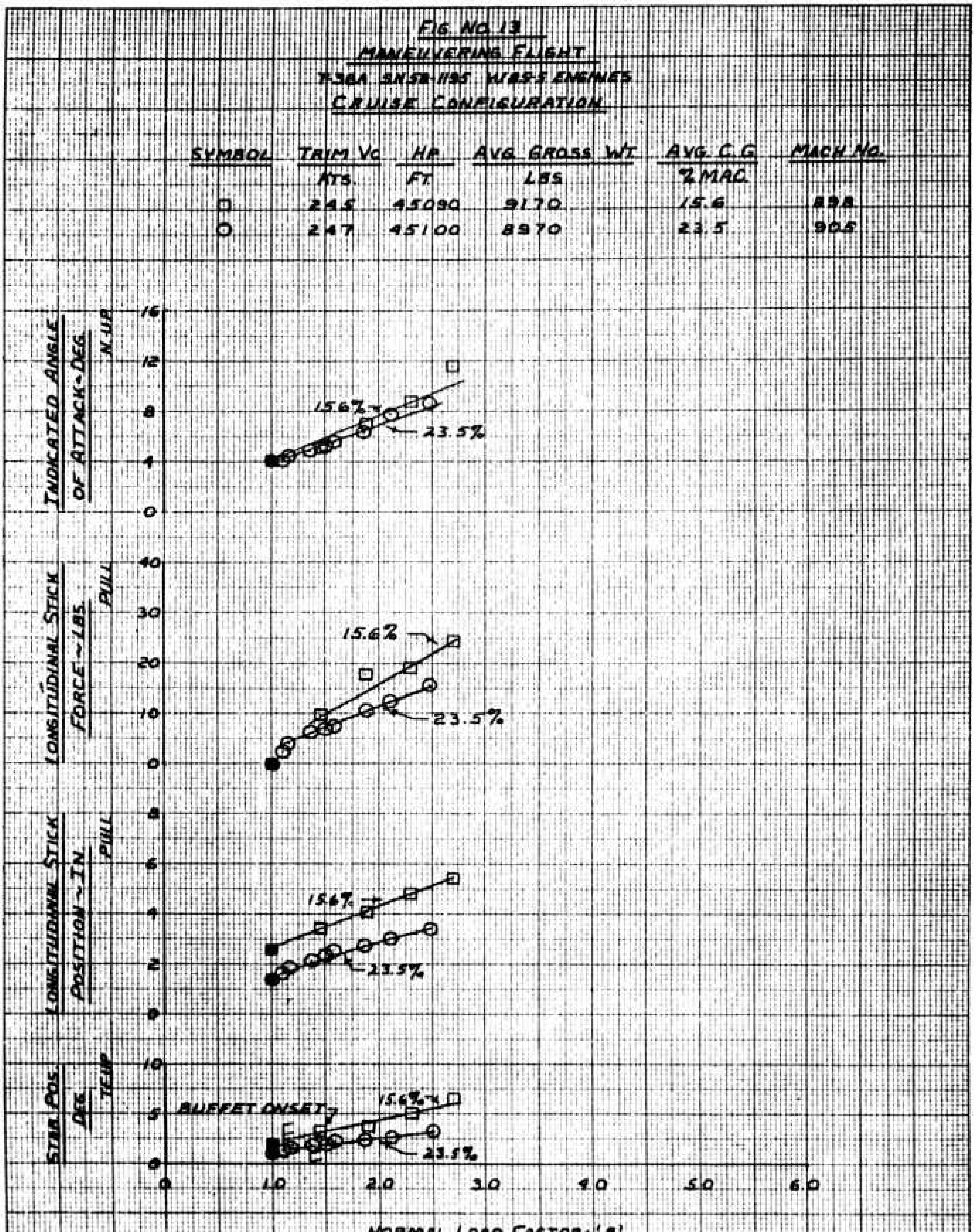


FIG. NO. 13
MANEUVERING FLIGHT
T-38A SN58-1195 W/J85-5 ENGINES
CRUISE CONFIGURATION
SYMBOL
TRIM Vc KTS.
Hp FT.
AVG. GROSS WT. LBS
AVG. C.G. % MAC
MACH NO.
□ 245 45090 9170 15.6 898
○ 247 45100 8970 23.5 905
INDICATED ANGLE OF ATTACK ~ DEG N.UP
LONGITUDINAL STICK FORCE ~ LBS PULL
LONGITUDINAL STICK POSITION ~ IN PULL
STAB. POS. DEG TEIP
15.6%
23.5%
BUFFET ONSET
NORMAL LOAD FACTOR ~ 'g'
0
1.0
2.0
3.0
4.0
5.0
6.0

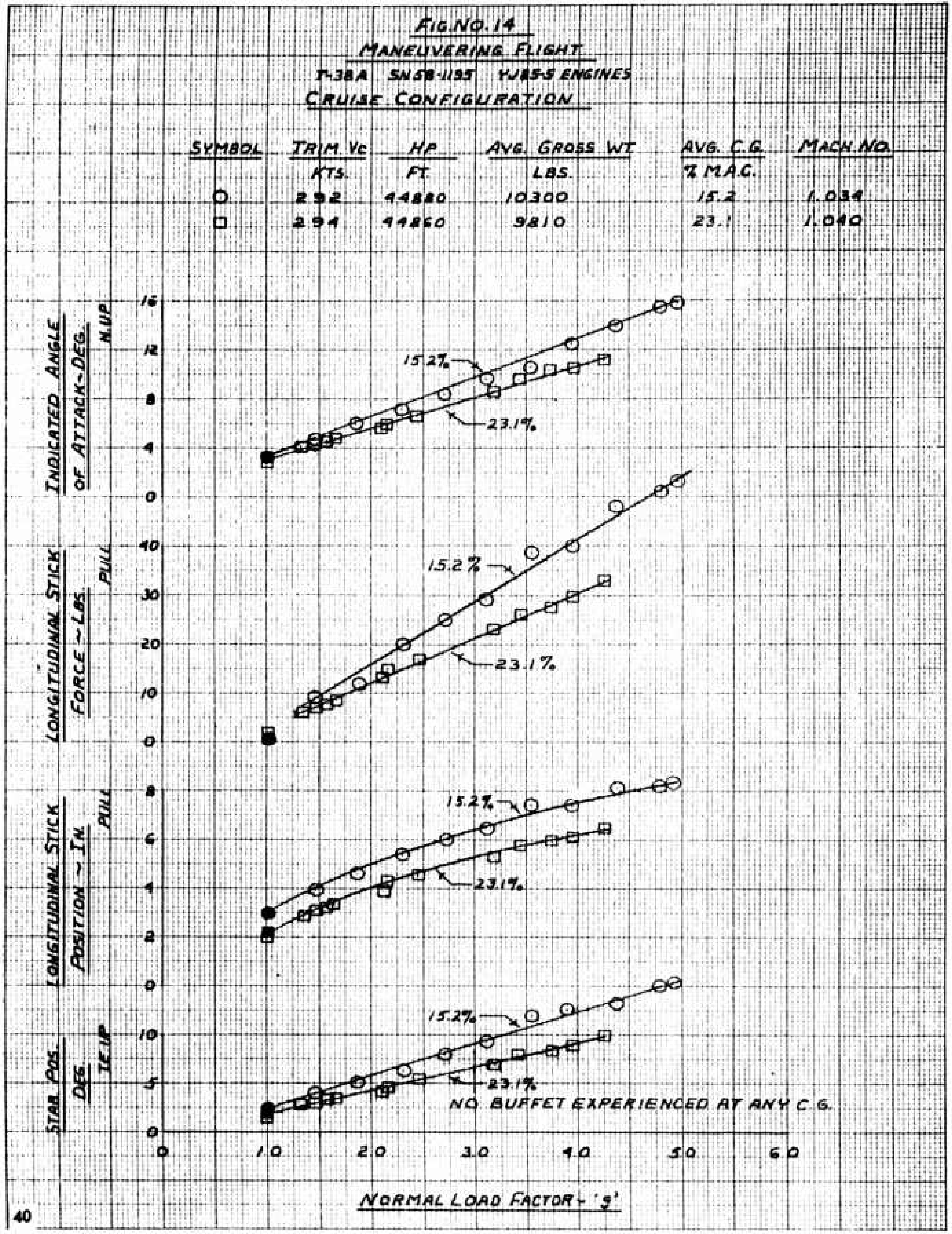


FIG. NO. 14
MANEUVERING FLIGHT
T-38A SN 58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION
SYMBOL | TRIM Vc KTS. | Hp FT. | AVG. GROSS WT LBS. | AVG. C.G. % M.A.C. | MACH NO.
○ | 292 | 44880 | 10300 | 15.2 | 1.034
□ | 294 | 44860 | 9810 | 23.1 | 1.040
INDICATED ANGLE OF ATTACK-DEG. N.UP
LONGITUDINAL STICK FORCE ~ LBS. PULL
LONGITUDINAL STICK POSITION ~ IN. PULL
STAB POS. DEG. TE.UP
15.2%
23.1%
NO BUFFET EXPERIENCED AT ANY C.G.
NORMAL LOAD FACTOR ~ 'g'

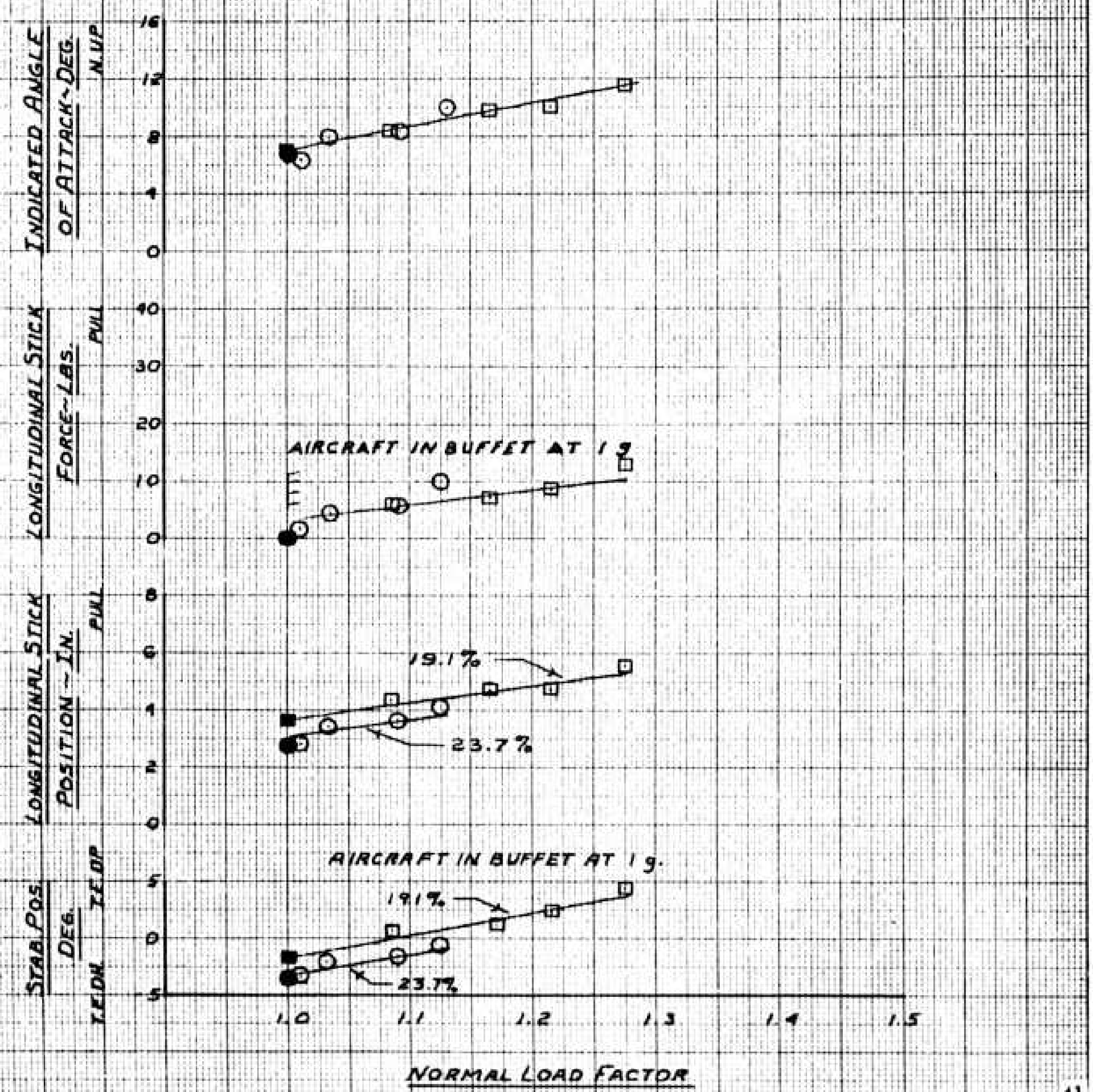
FIG. NO. 15

MANEUVERING FLIGHT

T-38A SN58-1195 YJ85-5 ENGINES

POWER APPROACH CONFIGURATION

SYMBOL	TRIM V_c KTS.	H_P FT.	AVG. GROSS WT. LBS.	AVG. C.G. % M.A.C.	MACH NO.
□	150	12580	8880	19.1	.287
○	149	10370	9030	23.7	.273



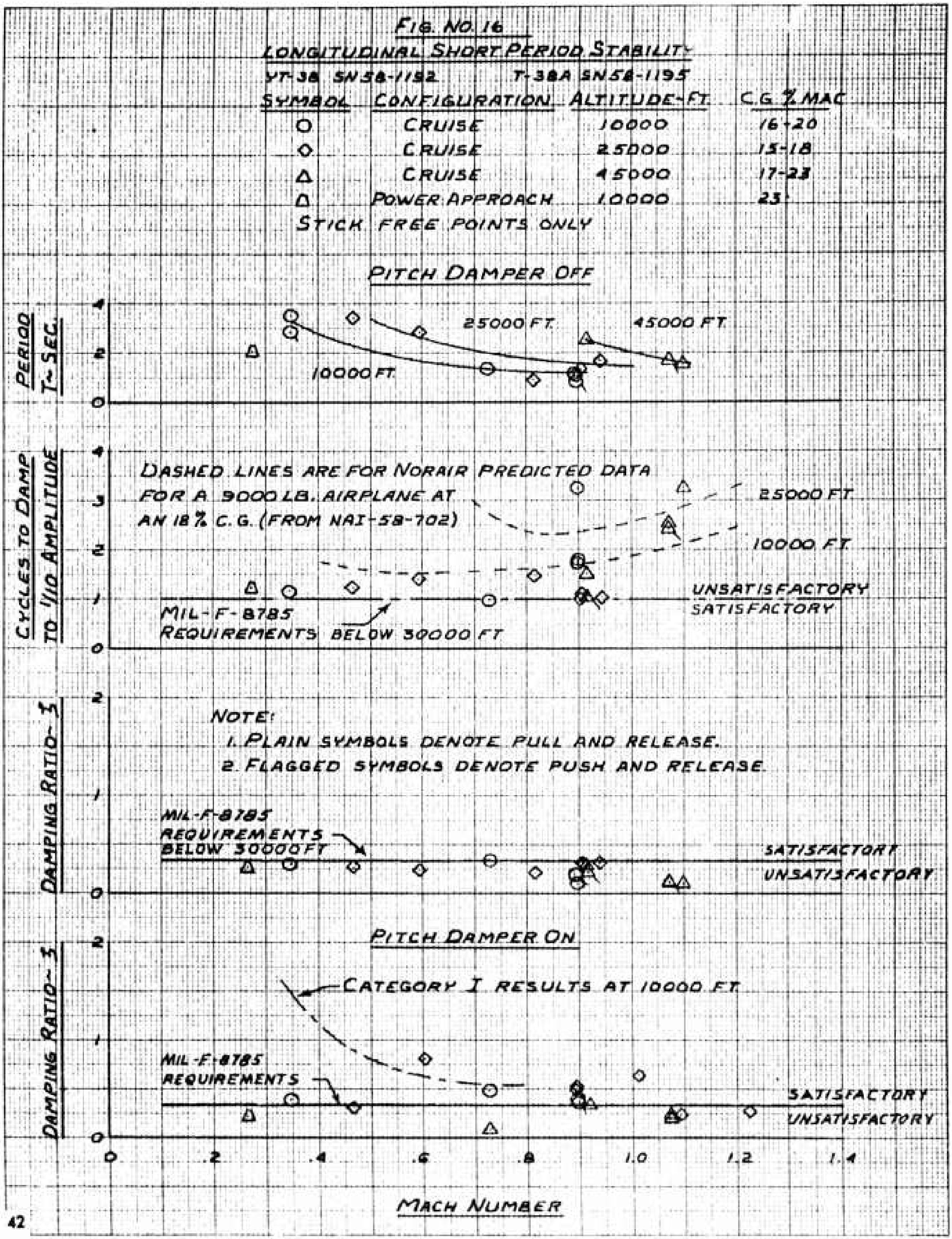


FIG. NO. 16
LONGITUDINAL SHORT PERIOD STABILITY
YT-38 SN58-1192 T-38A SN58-1195
SYMBOL CONFIGURATION ALTITUDE-FT. C.G. % MAC
CRUISE 10000 16-20
CRUISE 25000 15-18
CRUISE 45000 17-23
POWER APPROACH 10000 23
STICK FREE POINTS ONLY
PITCH DAMPER OFF
PERIOD T~SEC.
25000 FT.
45000 FT.
10000 FT.
CYCLES TO DAMP TO 1/10 AMPLITUDE
DASHED LINES ARE FOR NORAIR PREDICTED DATA FOR A 9000 LB. AIRPLANE AT AN 18% C.G. (FROM NAI-58-702)
25000 FT.
10000 FT.
UNSATISFACTORY
SATISFACTORY
MIL-F-8785 REQUIREMENTS BELOW 30000 FT
DAMPING RATIO~ζ
NOTE:
1. PLAIN SYMBOLS DENOTE PULL AND RELEASE.
2. FLAGGED SYMBOLS DENOTE PUSH AND RELEASE.
MIL-F-8785 REQUIREMENTS BELOW 30000 FT
SATISFACTORY
UNSATISFACTORY
PITCH DAMPER ON
DAMPING RATIO~ζ
CATEGORY I RESULTS AT 10000 FT.
MIL-F-8785 REQUIREMENTS
SATISFACTORY
UNSATISFACTORY
0 .2 .4 .6 .8 1.0 1.2 1.4
MACH NUMBER

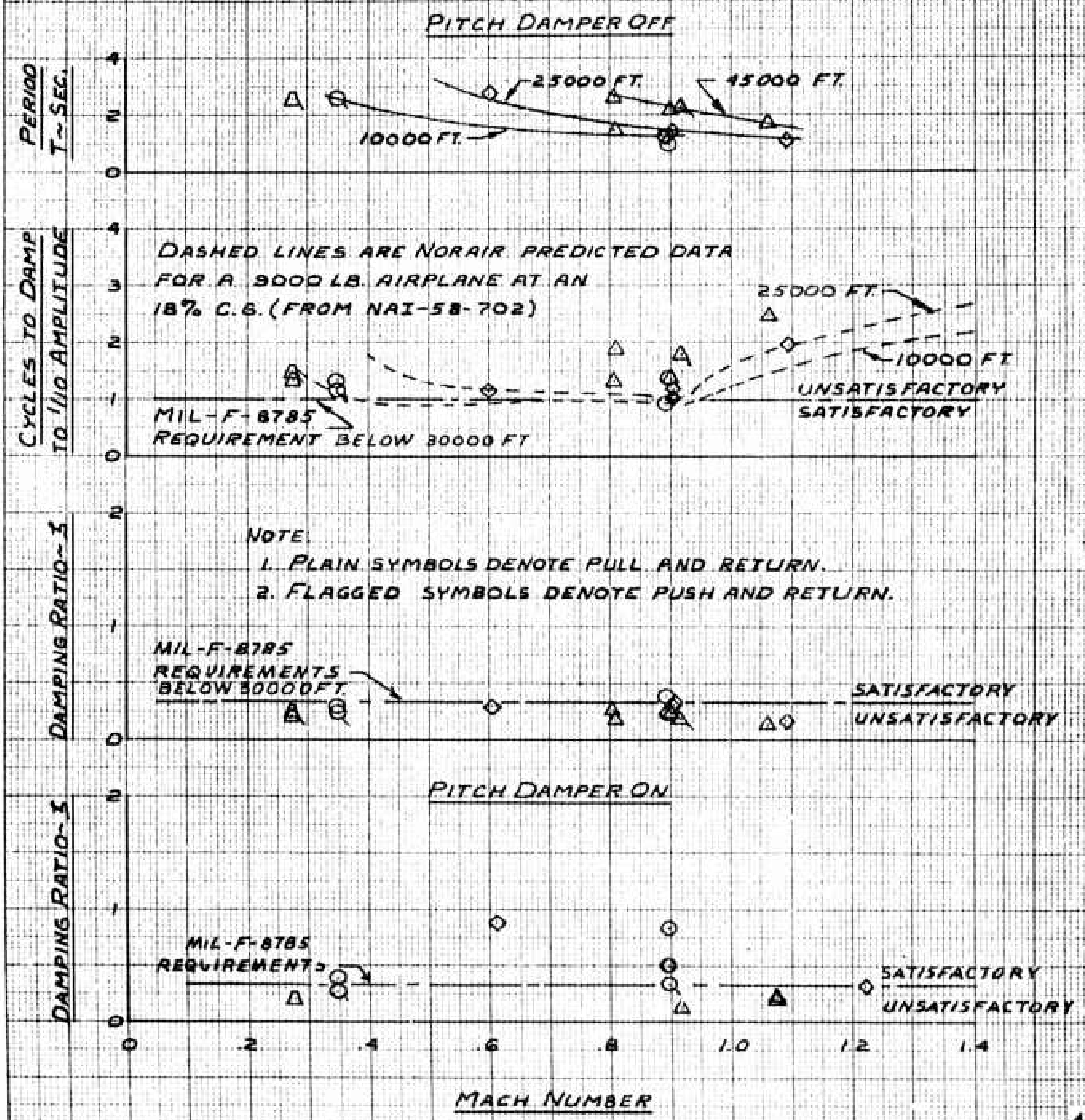
FIG. No. 17

LONGITUDINAL SHORT PERIOD STABILITY

YT-38 SN58-1192 T-38A SN58-1195

SYMBOL	CONFIGURATION	ALTITUDE~FT.	C.G.~% MAC
○	CRUISE	10000	16-20
◇	CRUISE	25000	15-18
△	CRUISE	45000	17-23
⊿	POWER APPROACH	10000	23

STICK FIXED POINTS ONLY



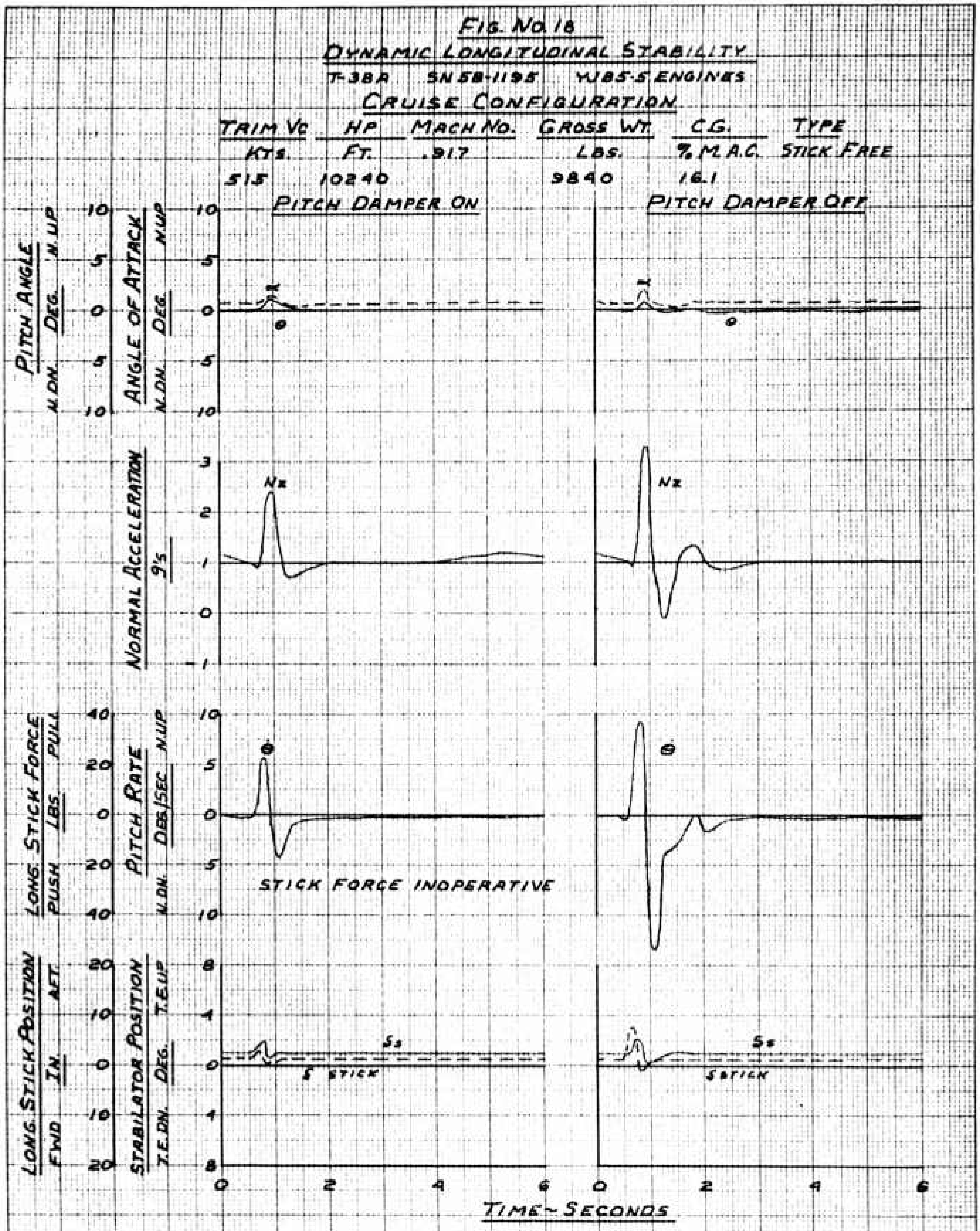


FIG. NO. 18
DYNAMIC LONGITUDINAL STABILITY
T-38A SN 58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION
TRIM Vc KTS. 515
HP FT. 10240
MACH NO. .917
GROSS WT. LBS. 9840
C.G. % M.A.C. 16.1
TYPE STICK FREE
PITCH DAMPER ON
PITCH DAMPER OFF
PITCH ANGLE N.DN. DEG. N.UP
ANGLE OF ATTACK N.DN. DEG. N.UP
NORMAL ACCELERATION g's
Nz
LONG. STICK FORCE PUSH LBS. PULL
PITCH RATE N.DN. DEG/SEC N.UP
STICK FORCE INOPERATIVE
LONG. STICK POSITION FWD IN. AFT
STABILATOR POSITION T.E.DN. DEG. T.E.UP
δs
δ STICK
TIME ~ SECONDS

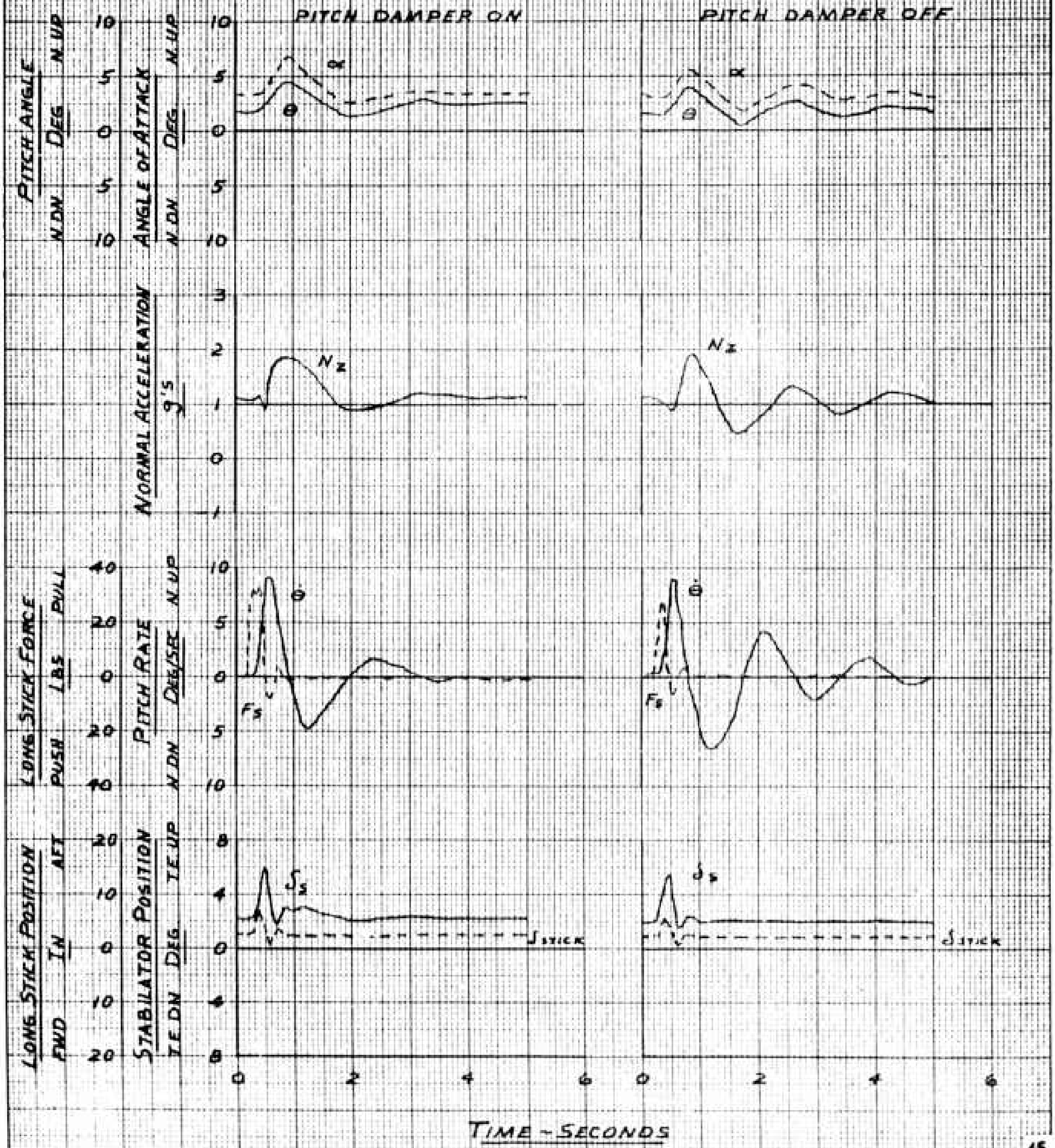
FIG. NO. 19

DYNAMIC LONGITUDINAL STABILITY

T-38A SN 58-1195 YJ85-5 ENGINES

CRUISE CONFIGURATION

TRIM V_c KTS	H_p FT	MACH NO	GROSS WT LBS	C.G %MAC	TYPE
302	45290	1.075	10250	22.5	STICK FREE



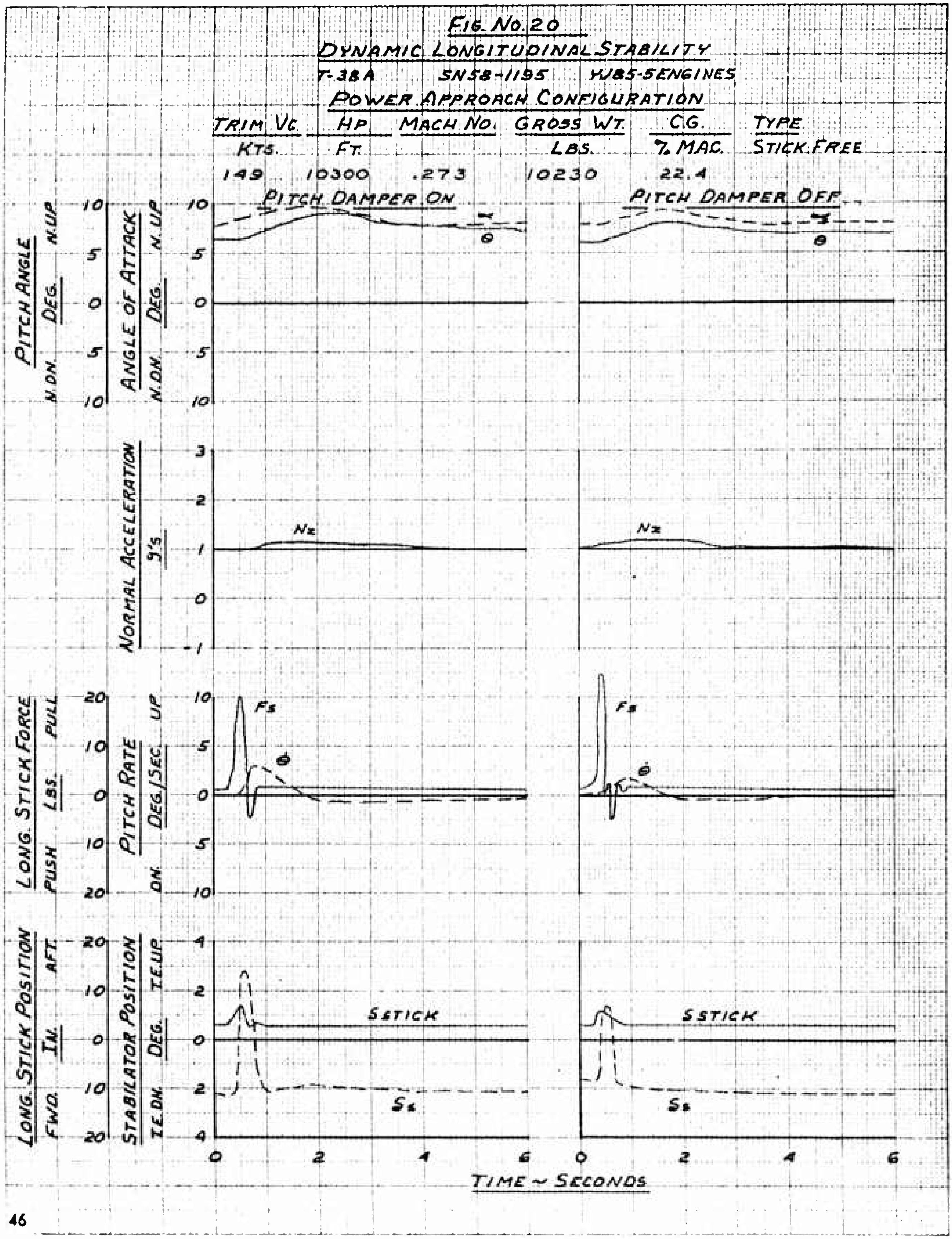


FIG. NO. 20
DYNAMIC LONGITUDINAL STABILITY
T-38A SN58-1195 YJ85-5 ENGINES
POWER APPROACH CONFIGURATION
TRIM VC KTS. 149
HP FT. 10300
MACH NO. .273
GROSS WT. LBS. 10230
C.G. % MAC. 22.4
TYPE STICK FREE
PITCH DAMPER ON
PITCH DAMPER OFF
PITCH ANGLE DEG. N.UP N.DN.
ANGLE OF ATTACK DEG. N.UP N.DN.
NORMAL ACCELERATION g's
Nz
LONG. STICK FORCE LBS. PULL PUSH
PITCH RATE DEG./SEC. UP DN.
Fs
LONG. STICK POSITION IN. AFT. FWD.
STABILATOR POSITION DEG. T.E.UP T.E.DN.
δ STICK
δs
TIME ~ SECONDS

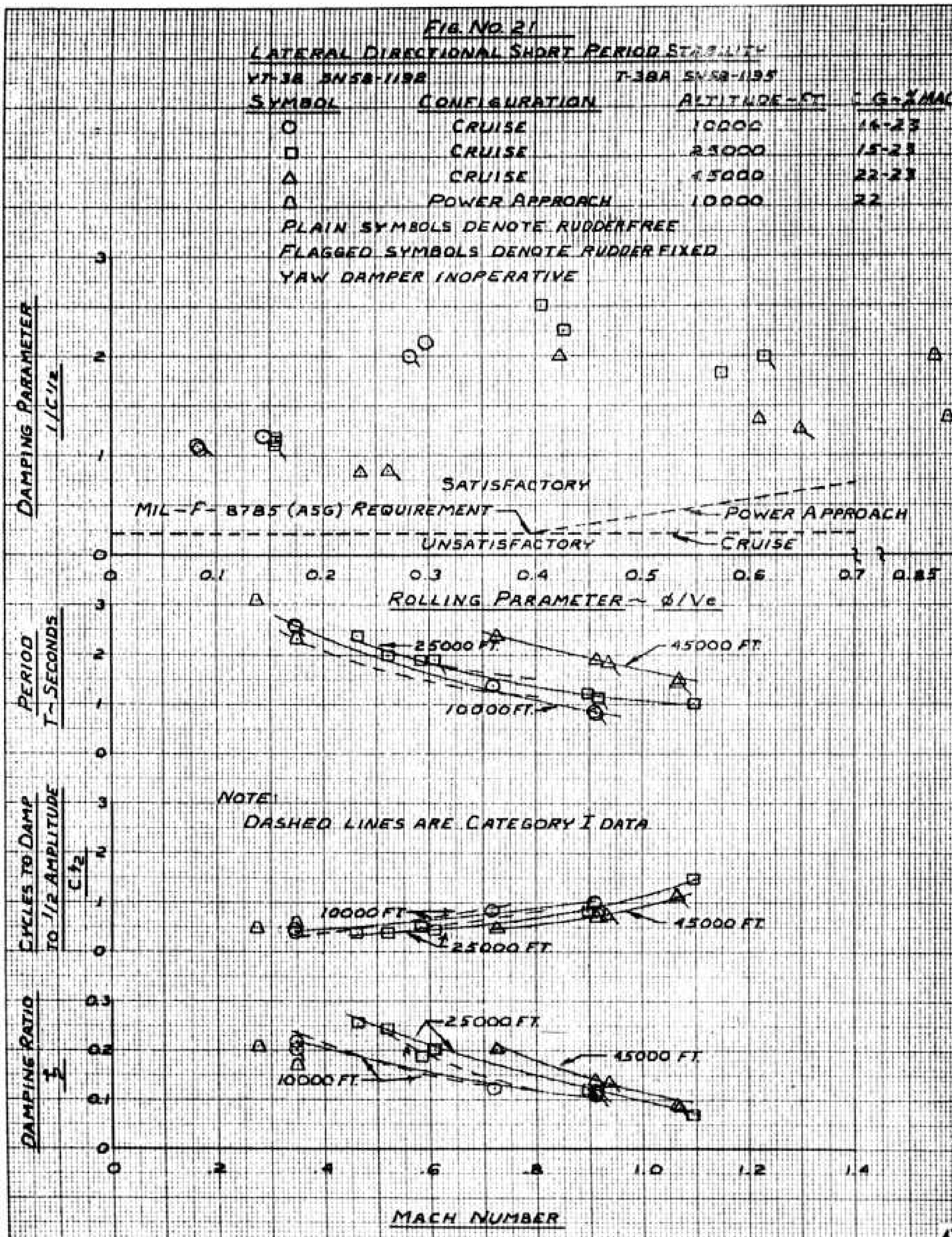


FIG. NO 21
LATERAL DIRECTIONAL SHORT PERIOD
YT-38 SN58-1192
T-38A SN58-1195
SYMBOL
CONFIGURATION
ALTITUDE-FT
C.G.-% MAC
CRUISE
10000
14-23
CRUISE
25000
15-23
CRUISE
45000
22-23
POWER APPROACH
10000
22
PLAIN SYMBOLS DENOTE RUDDERFREE
FLAGGED SYMBOLS DENOTE RUDDER FIXED
YAW DAMPER INOPERATIVE
DAMPING PARAMETER
1/C½
SATISFACTORY
MIL-F-8785 (ASG) REQUIREMENT
POWER APPROACH
UNSATISFACTORY
CRUISE
ROLLING PARAMETER ~ ϕ/Ve
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.85
PERIOD
T-SECONDS
25000 FT.
45000 FT.
10000 FT.
NOTE:
DASHED LINES ARE CATEGORY I DATA
CYCLES TO DAMP
TO 1/2 AMPLITUDE
C½
10000 FT.
45000 FT.
25000 FT.
DAMPING RATIO
25000 FT.
45000 FT.
10000 FT.
MACH NUMBER
.2
.4
.6
.8
1.0
1.2
1.4

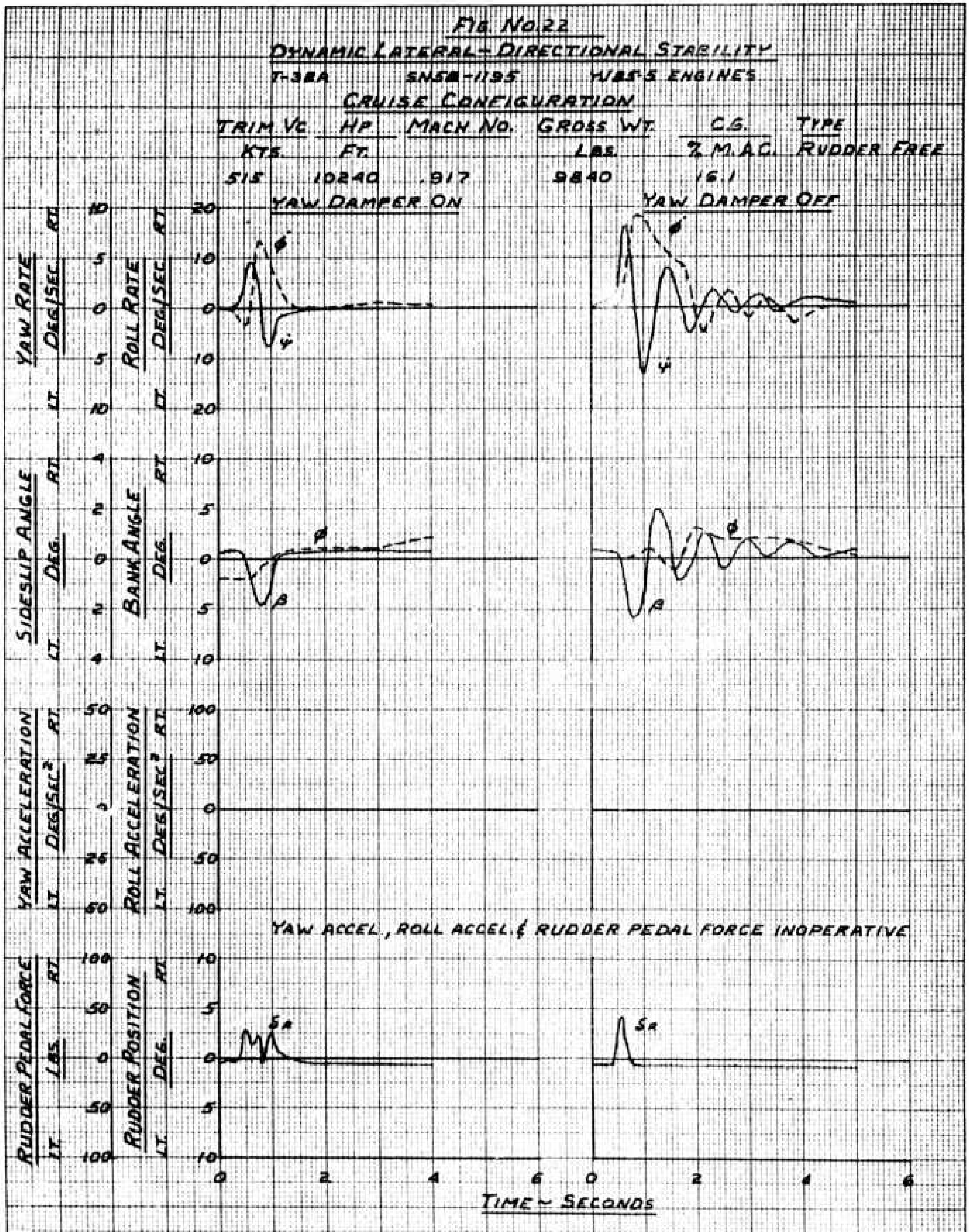


FIG. NO. 22
DYNAMIC LATERAL-DIRECTIONAL STABILITY
T-38A SN58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION
TRIM Vc KTS. HP FT. MACH NO. GROSS WT. LBS. C.G. % M.A.C. TYPE
515 10240 .917 9840 16.1 RUDDER FREE
YAW DAMPER ON
YAW DAMPER OFF
YAW RATE DEG/SEC LT. RT.
ROLL RATE DEG/SEC LT. RT.
SIDESLIP ANGLE DEG. LT. RT.
BANK ANGLE DEG. LT. RT.
YAW ACCELERATION DEG/SEC² LT. RT.
ROLL ACCELERATION DEG/SEC² LT. RT.
YAW ACCEL., ROLL ACCEL. & RUDDER PEDAL FORCE INOPERATIVE
RUDDER PEDAL FORCE LBS. LT. RT.
RUDDER POSITION DEG. LT. RT.
TIME ~ SECONDS

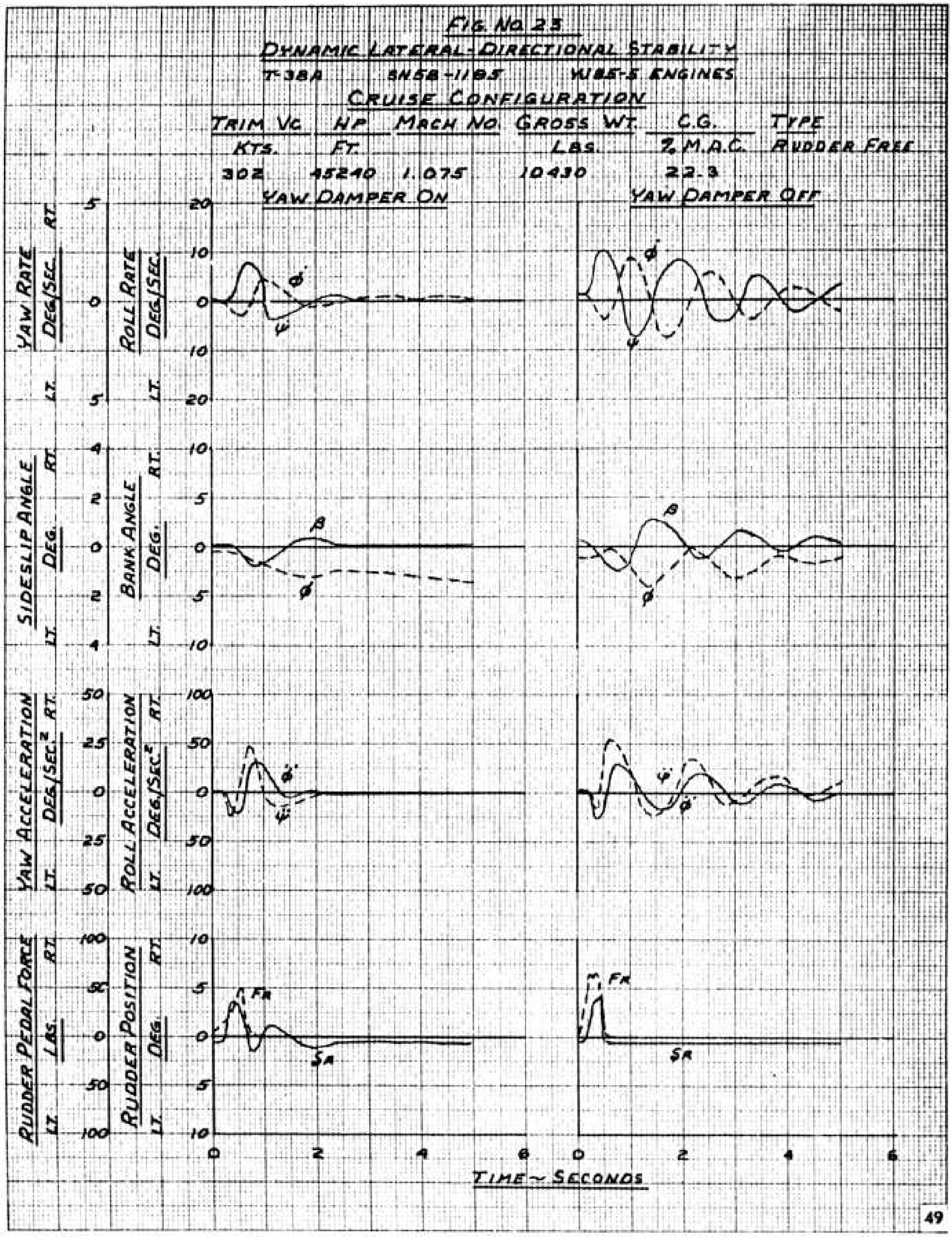


FIG. NO. 23
DYNAMIC LATERAL-DIRECTIONAL STABILITY
T-38A 8N58-1185 J85-5 ENGINES
CRUISE CONFIGURATION
TRIM Vc KTS. 302
HP FT. 45240
MACH NO. 1.075
GROSS WT. LBS. 10430
C.G. % M.A.C. 22.3
TYPE RUDDER FREE
YAW DAMPER ON
YAW DAMPER OFF
YAW RATE DEG/SEC. LT. RT.
ROLL RATE DEG/SEC. LT. RT.
SIDESLIP ANGLE DEG. LT. RT.
BANK ANGLE DEG. LT. RT.
YAW ACCELERATION DEG/SEC.² LT. RT.
ROLL ACCELERATION DEG/SEC.² LT. RT.
RUDDER PEDAL FORCE LBS. LT. RT.
RUDDER POSITION DEG. LT. RT.
TIME ~ SECONDS

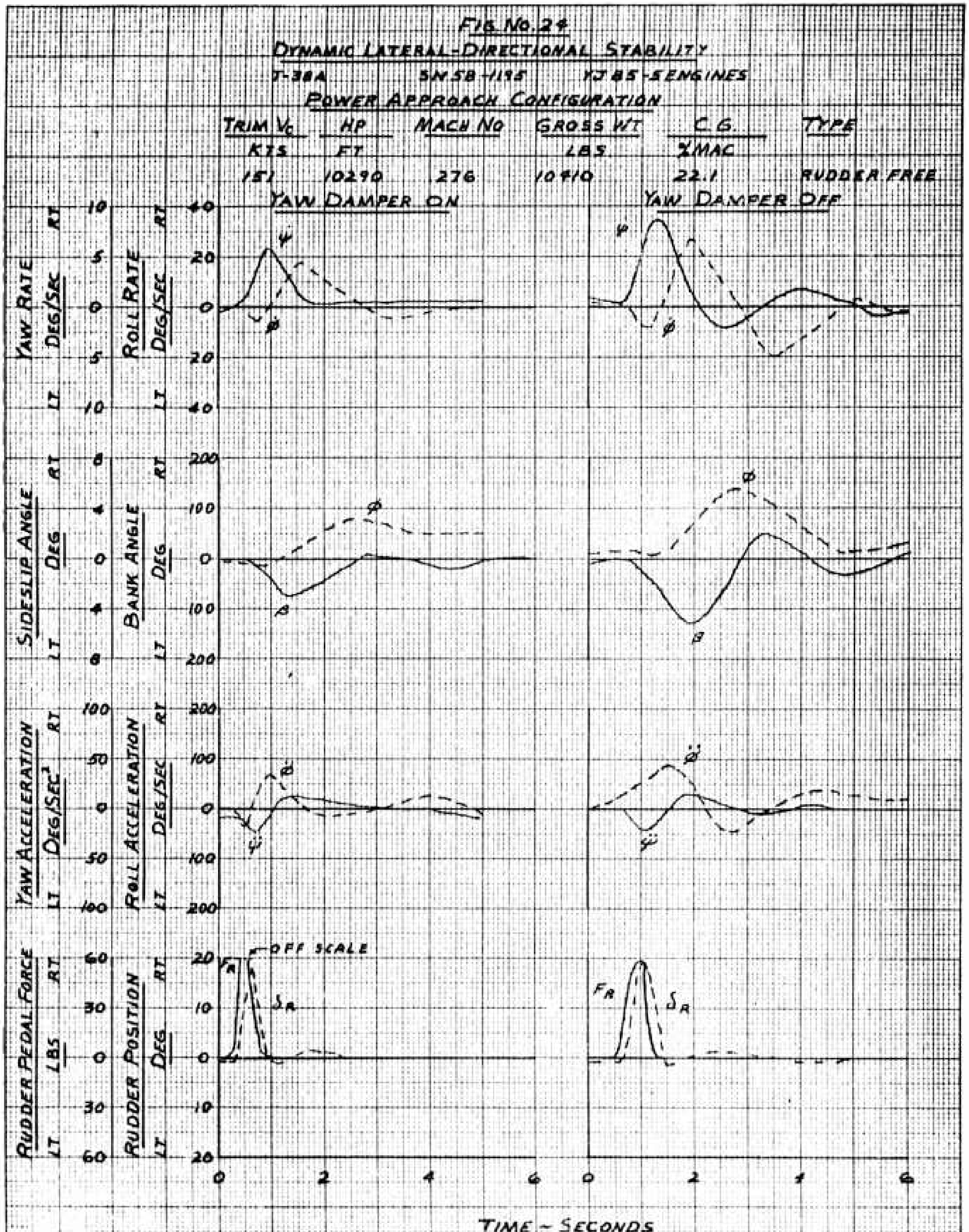


FIG NO. 24
DYNAMIC LATERAL-DIRECTIONAL STABILITY
T-38A SN 58-1195 YJ 85-5 ENGINES
POWER APPROACH CONFIGURATION
TRIM V_C KTS 151
HP FT 10290
MACH NO .276
GROSS WT LBS 10410
C.G. %MAC 22.1
TYPE RUDDER FREE
YAW DAMPER ON
YAW DAMPER OFF
YAW RATE DEG/SEC
ROLL RATE DEG/SEC
SIDESLIP ANGLE DEG
BANK ANGLE DEG
YAW ACCELERATION DEG/SEC²
ROLL ACCELERATION DEG/SEC
RUDDER PEDAL FORCE LBS
RUDDER POSITION DEG
RT
LT
OFF SCALE
F_R
δ_R
TIME ~ SECONDS

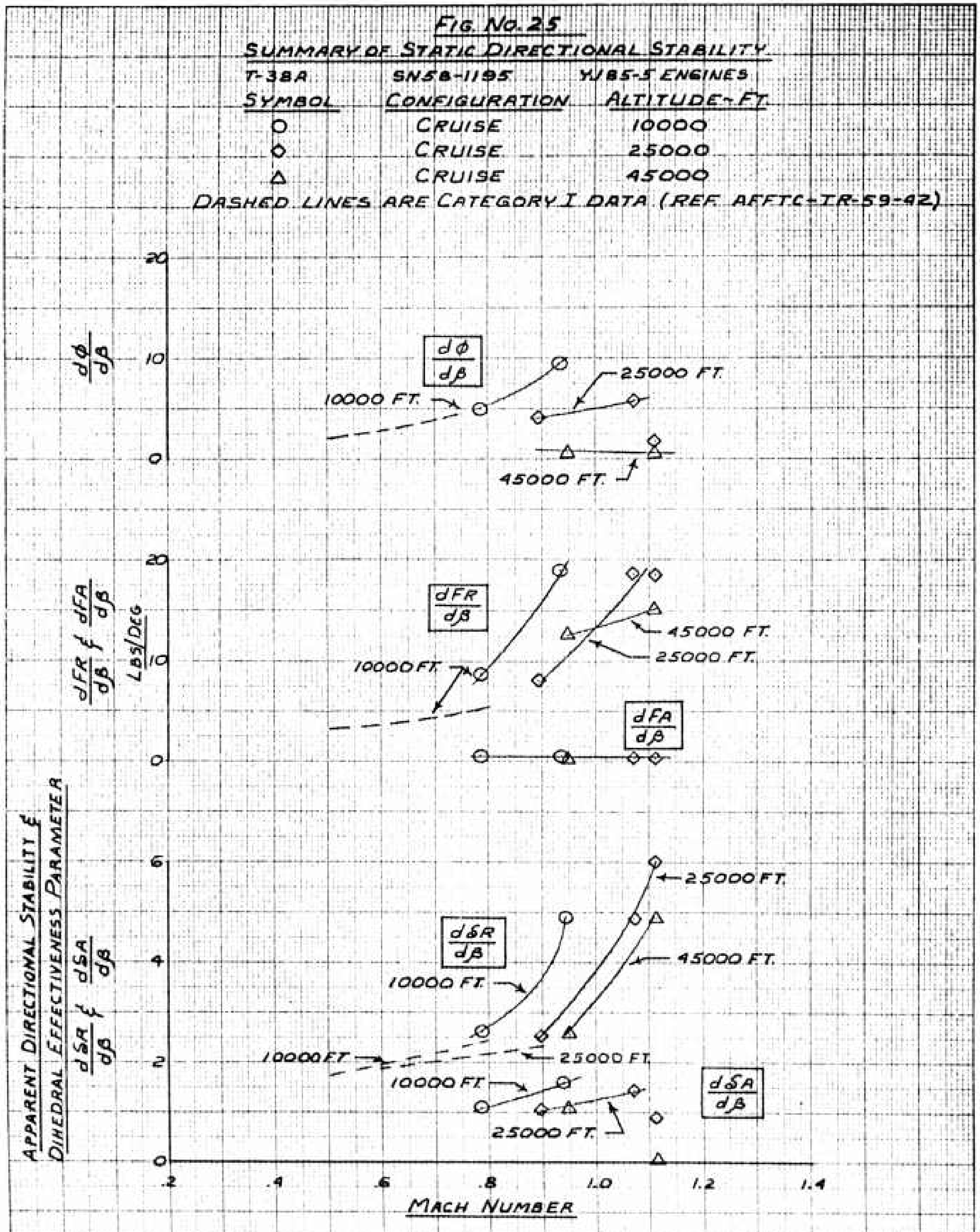


FIG. NO. 25
SUMMARY OF STATIC DIRECTIONAL STABILITY
T-38A
SN58-1195
YJ85-5 ENGINES
SYMBOL
CONFIGURATION
ALTITUDE ~ FT
CRUISE
10000
CRUISE
25000
CRUISE
45000
DASHED LINES ARE CATEGORY I DATA (REF. AFFTC-TR-59-42)
$\frac{d\phi}{d\beta}$
10000 FT.
25000 FT.
45000 FT.
$\frac{dF_R}{d\beta}$ & $\frac{dF_A}{d\beta}$ LBS/DEG
$\frac{dF_R}{d\beta}$
$\frac{dF_A}{d\beta}$
APPARENT DIRECTIONAL STABILITY & DIHEDRAL EFFECTIVENESS PARAMETER
$\frac{d\delta_R}{d\beta}$ & $\frac{d\delta_A}{d\beta}$
$\frac{d\delta_R}{d\beta}$
$\frac{d\delta_A}{d\beta}$
MACH NUMBER
.2
.4
.6
.8
1.0
1.2
1.4
0
2
4
6
10
20

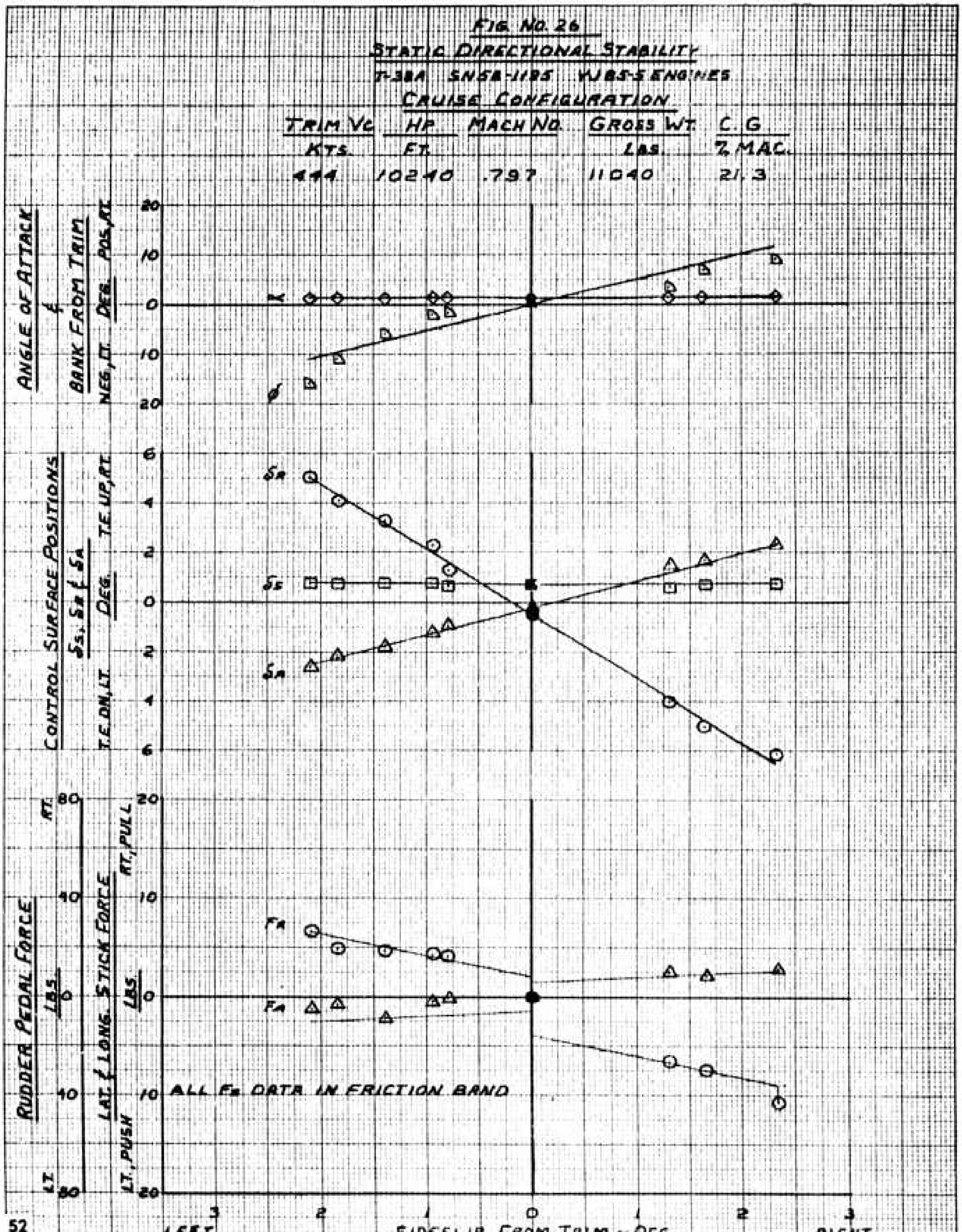


FIG. NO. 26
STATIC DIRECTIONAL STABILITY
T-38A SN58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION
TRIM Vc KTS. 444
HP FT. 10240
MACH NO. .797
GROSS WT. LBS. 11040
C.G % MAC. 21.3
ANGLE OF ATTACK & BANK FROM TRIM
NEG., LT. DEG. POS., RT.
CONTROL SURFACE POSITIONS
δS, δR & δA
T.E. DN, LT. DEG. T.E. UP, RT.
RUDDER PEDAL FORCE LBS.
LAT. & LONG. STICK FORCE LBS.
RT., PULL
LT., PUSH
ALL FS DATA IN FRICTION BAND
LEFT
SIDESLIP FROM TRIM ~ DEG
RIGHT

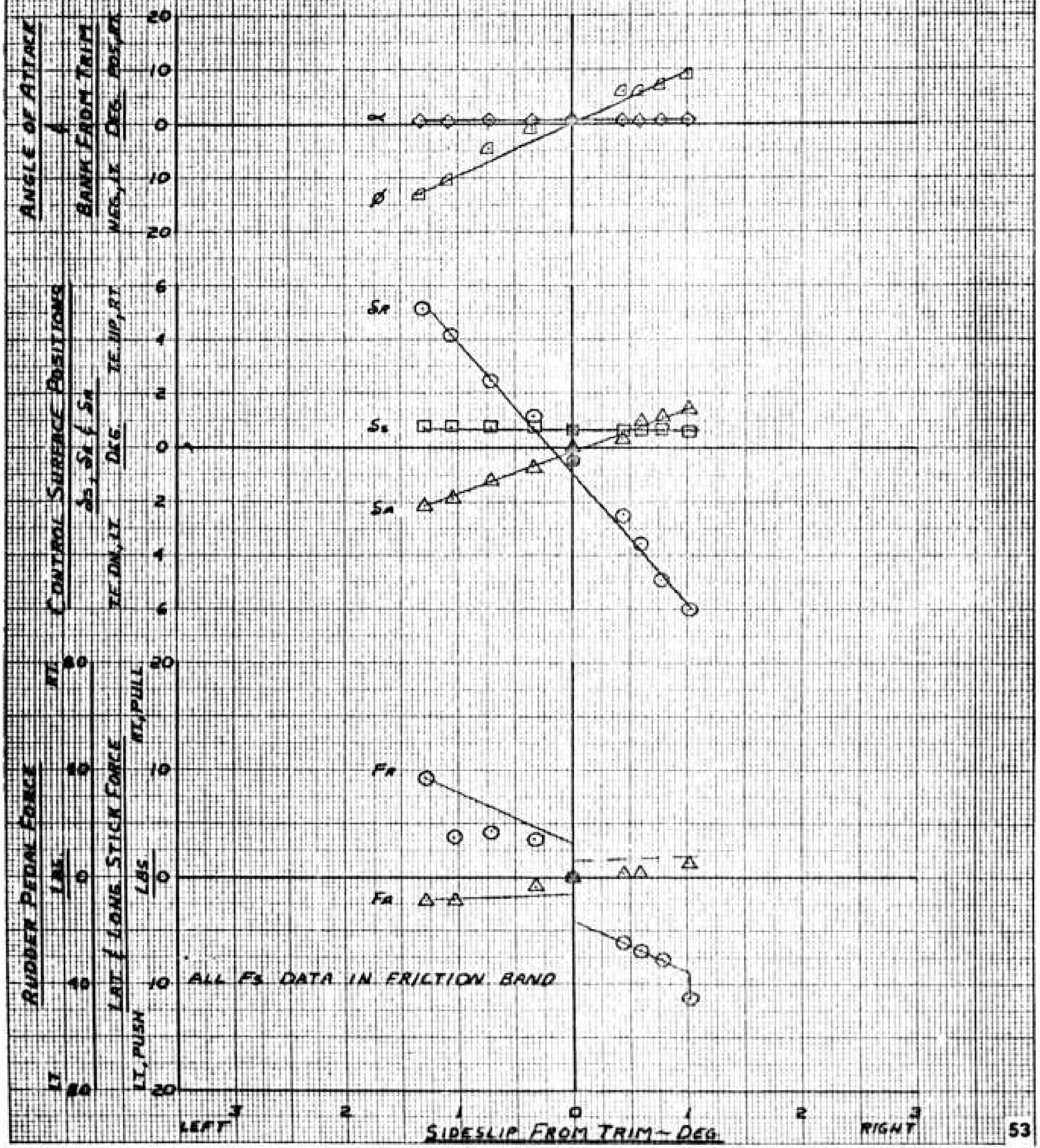
FIG. NO. 27

STATIC DIRECTIONAL STABILITY

T-38A SN 58-1195 YJ85-5 ENGINES

CRUISE CONFIGURATION

TRIM V_C KTS.	HP FT.	MACH NO.	GROSS WT LBS	C.G. % MAC.
530	9580	.931	11400	21.3



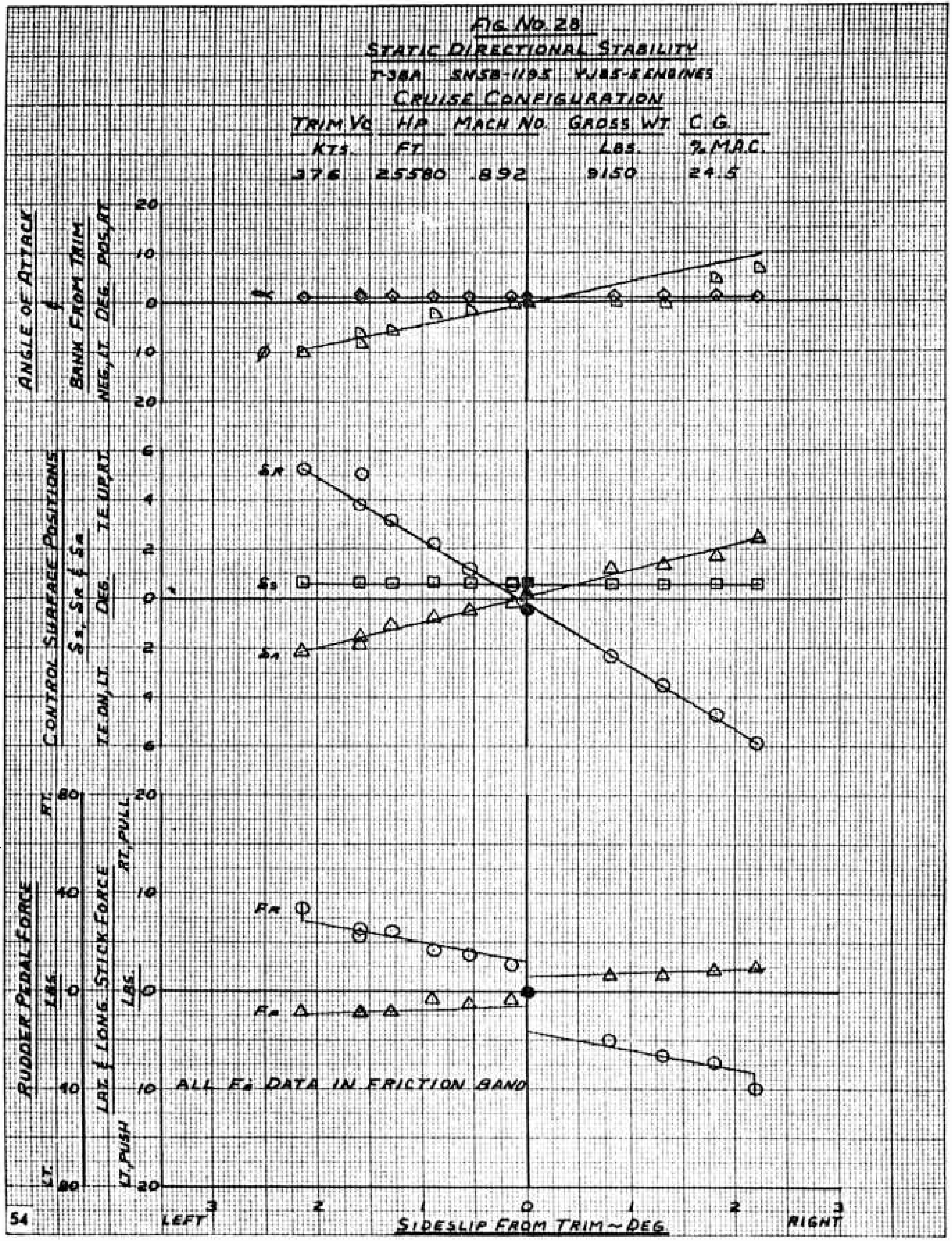


FIG. NO. 28
STATIC DIRECTIONAL STABILITY
T-38A SN58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION
TRIM V_c KTS. 376
HP FT. 25580
MACH NO. .892
GROSS WT LBS. 9150
C.G. % M.A.C. 24.5
ANGLE OF ATTACK & BANK FROM TRIM
NEG, LT. DEG. POS, RT.
CONTROL SURFACE POSITIONS δ_s, δ_R & δ_A
T.E. DN, LT. DEG. T.E. UP, RT.
RUDDER PEDAL FORCE LBS. LT. RT.
LAT & LONG STICK FORCE LBS. LT, PUSH RT, PULL
α
ϕ
δ_R
δ_s
δ_A
F_R
F_A
ALL F_A DATA IN FRICTION BAND
LEFT
RIGHT
SIDESLIP FROM TRIM ~ DEG.

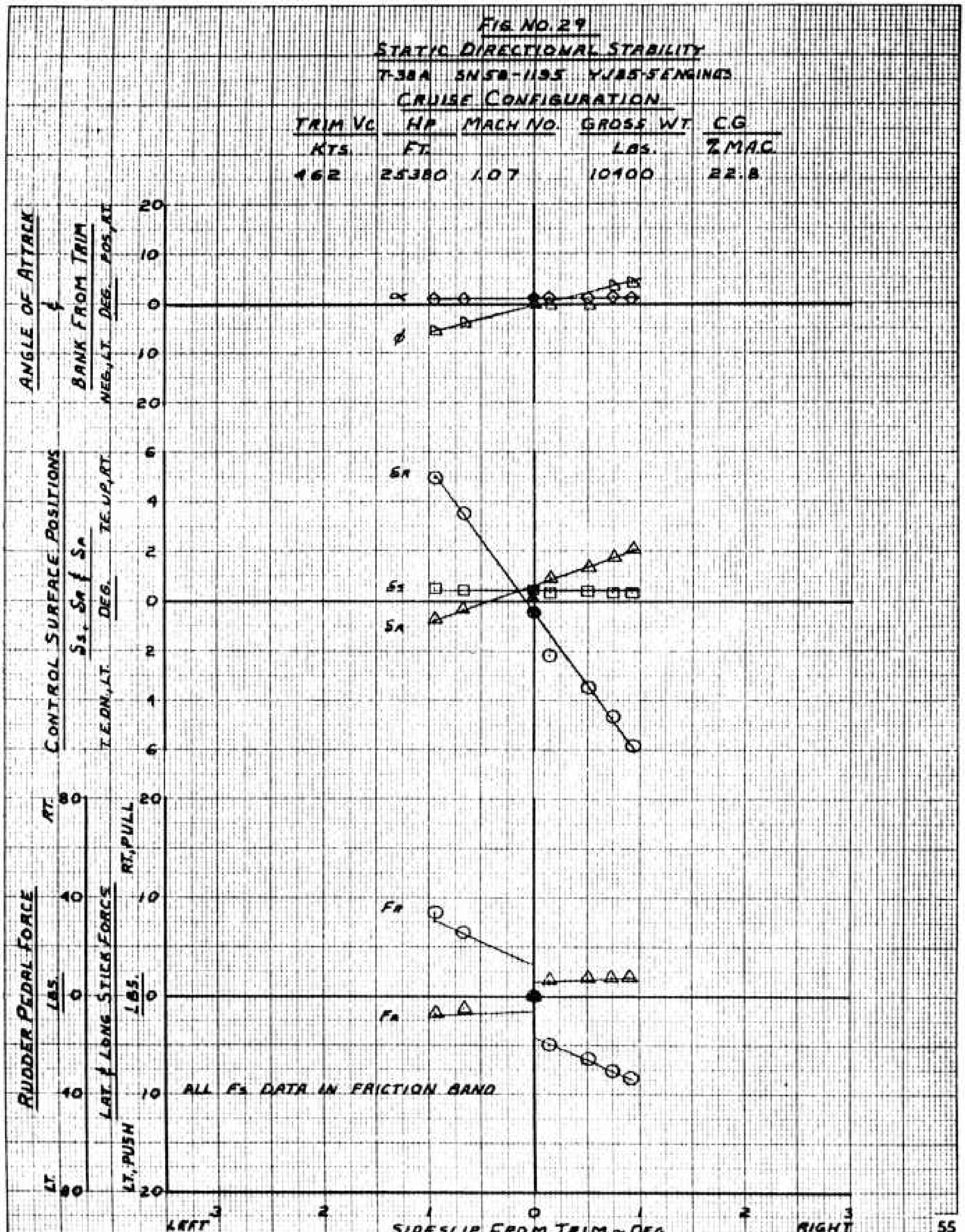


FIG. NO. 29
STATIC DIRECTIONAL STABILITY
T-38A SN 58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION
TRIM Vc KTS. 462
HP FT. 25380
MACH NO. 1.07
GROSS WT LBS. 10400
C.G % M.A.C. 22.8
ANGLE OF ATTACK & BANK FROM TRIM
NEG., LT. DEG. POS., RT.
CONTROL SURFACE POSITIONS
T.E. DN., LT. DEG. T.E. UP, RT.
RUDDER PEDAL FORCE LBS.
LT. RT.
LAT & LONG STICK FORCE LBS.
LT., PUSH RT., PULL
ALL Fs DATA IN FRICTION BAND
LEFT
RIGHT
SIDESLIP FROM TRIM ~ DEG.

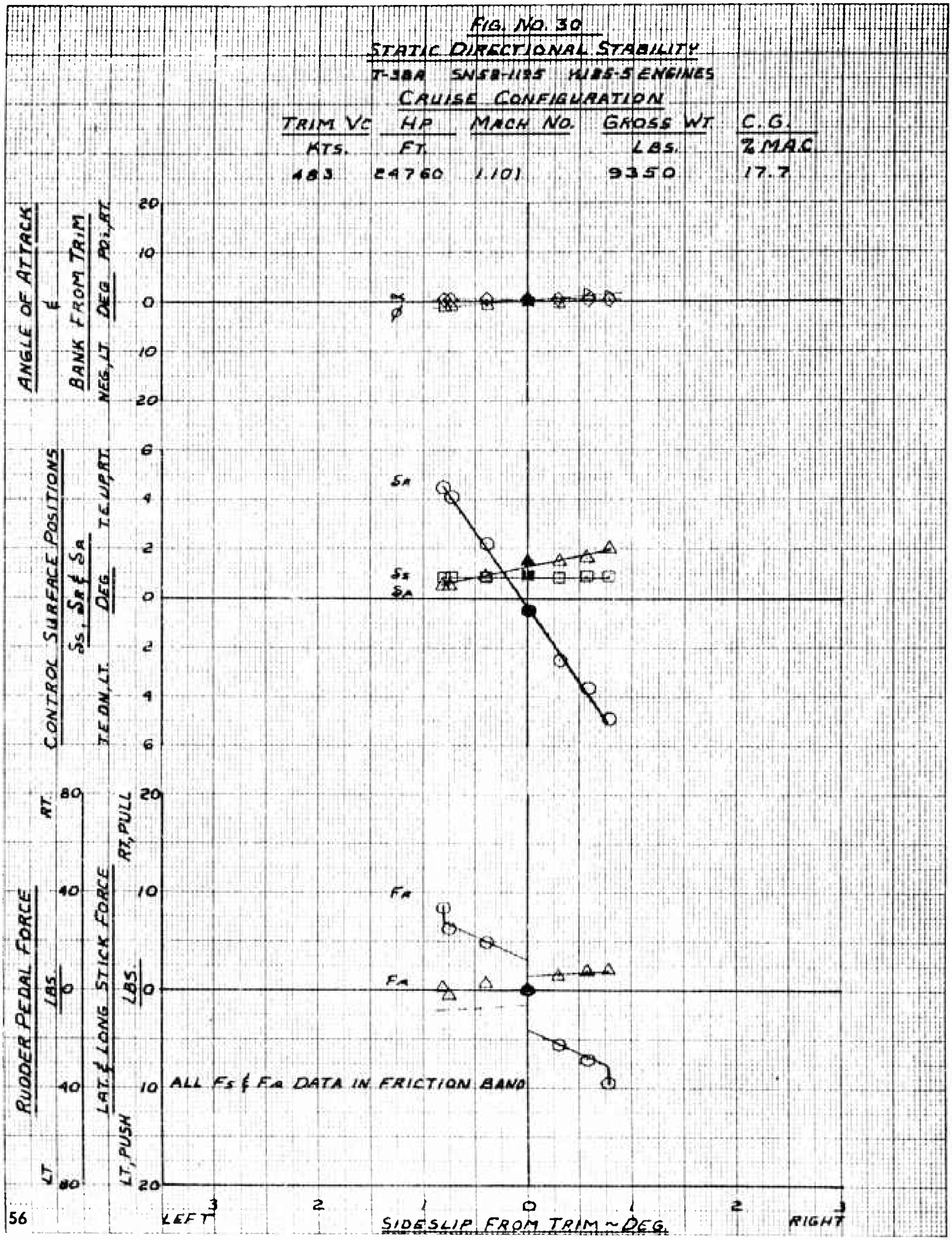


FIG. NO. 30
STATIC DIRECTIONAL STABILITY
T-38A SN58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION
TRIM Vc KTS. 483
HP FT. 24760
MACH NO. 1.101
GROSS WT LBS. 9350
C.G. % MAC 17.7
ANGLE OF ATTACK & BANK FROM TRIM NEG, LT DEG POS, RT
CONTROL SURFACE POSITIONS δ_S, δ_R & δ_A DEG T.E. UP, RT T.E. DN, LT.
RUDDER PEDAL FORCE LBS. RT. LT
LAT. & LONG. STICK FORCE LBS. RT, PULL LT, PUSH
ALL Fs & Fa DATA IN FRICTION BAND
LEFT
RIGHT
SIDESLIP FROM TRIM ~ DEG.

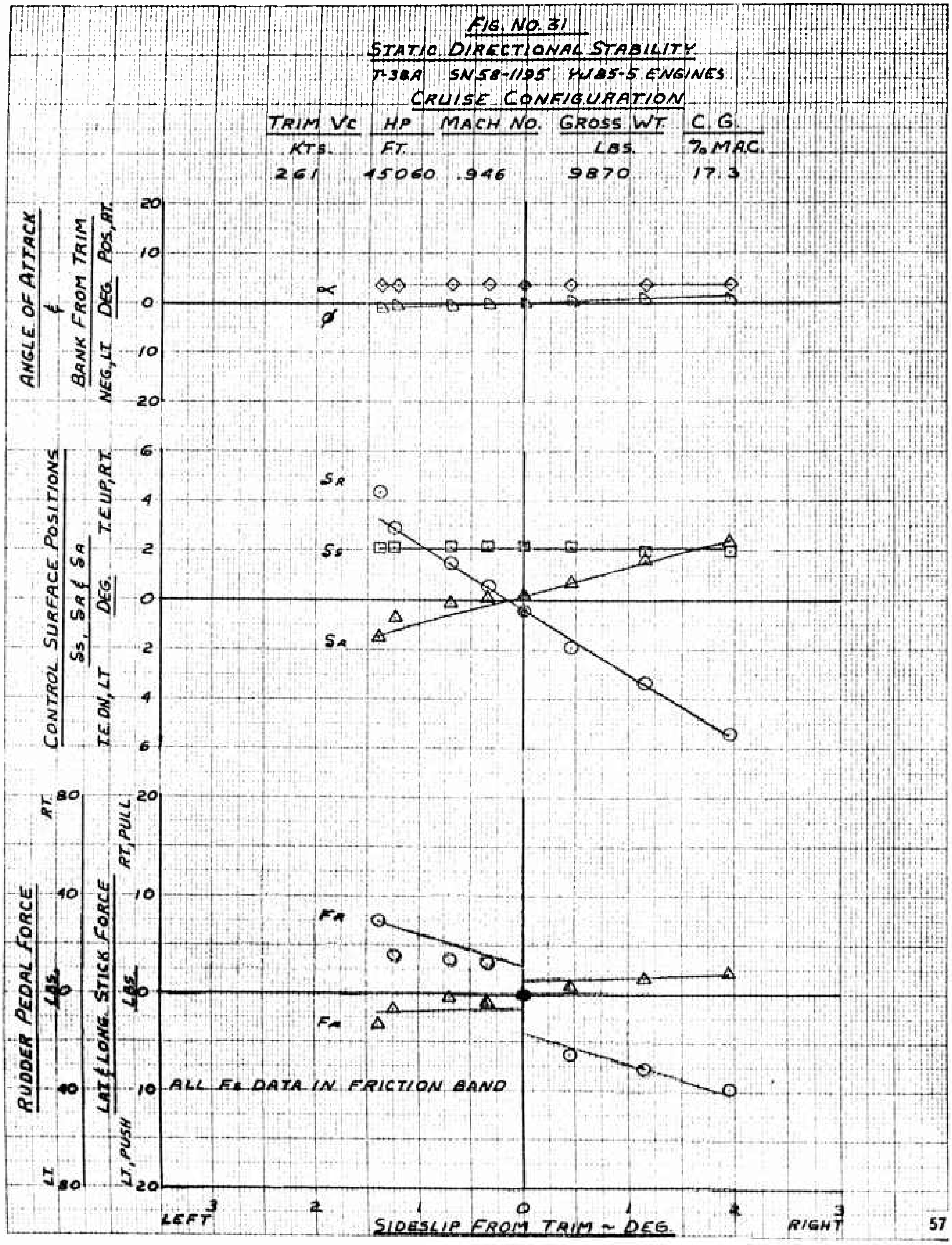


FIG. NO. 31
STATIC DIRECTIONAL STABILITY
T-38A SN 58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION
TRIM Vc KTS. 261
HP FT. 45060
MACH NO. .946
GROSS WT. LBS. 9870
C.G. % MAC 17.3
ANGLE OF ATTACK & BANK FROM TRIM
NEG, LT DEG. POS, RT.
CONTROL SURFACE POSITIONS
δs, δR & δA
T.E. DN, LT DEG. T.E. UP, RT.
RUDDER PEDAL FORCE LBS.
LAT & LONG. STICK FORCE LBS.
RT, PULL
LT, PUSH
ALL FA DATA IN FRICTION BAND
LEFT
RIGHT
SIDESLIP FROM TRIM ~ DEG.

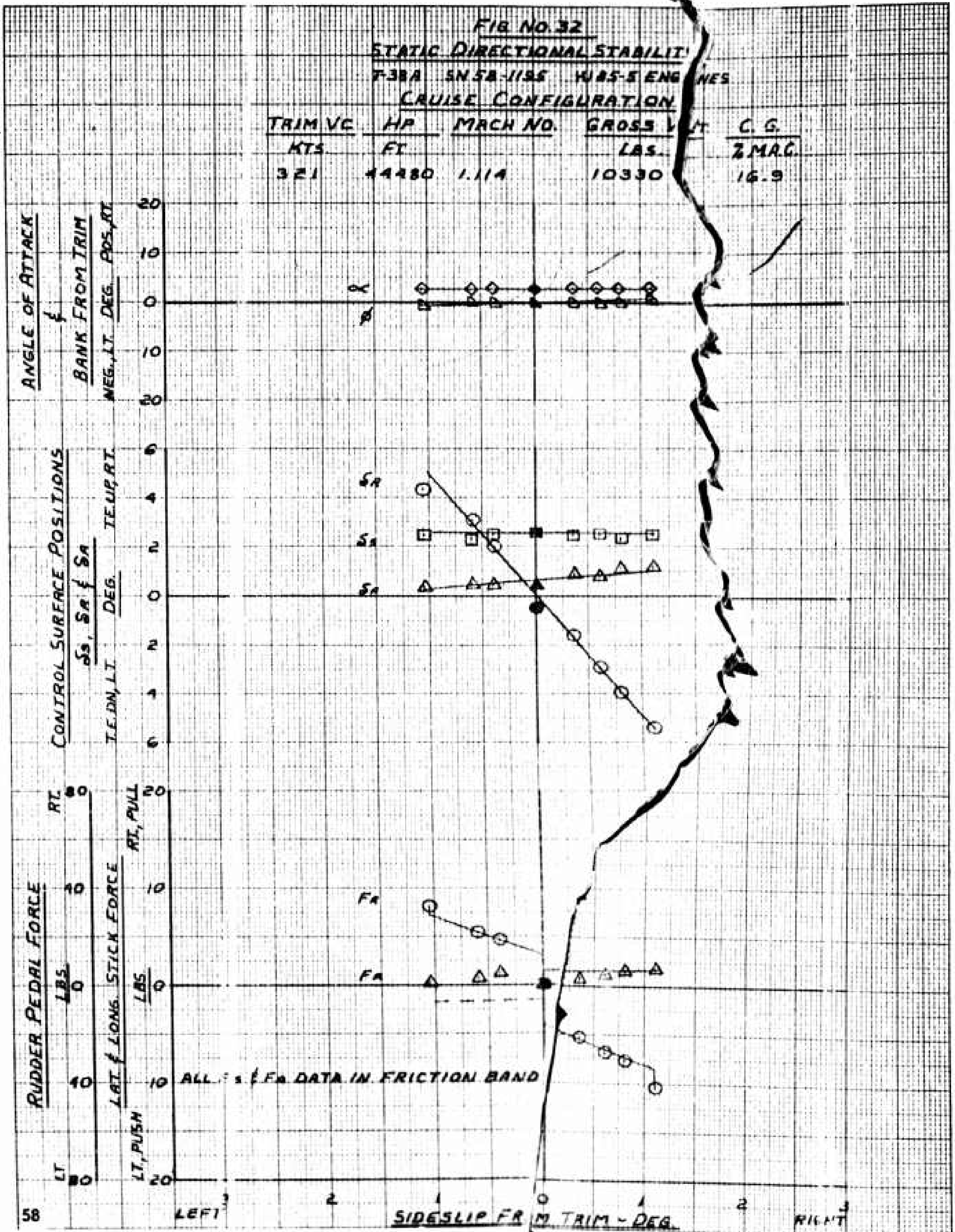


FIG NO. 32
STATIC DIRECTIONAL STABILITY
T-38A SN 58-1195 J85-5 ENGINES
CRUISE CONFIGURATION
TRIM VC KTS 321
HP FT 44480
MACH NO. 1.114
GROSS WT LBS. 10330
C. G. % MAC 16.9
ANGLE OF ATTACK & BANK FROM TRIM
NEG., LT. DEG. POS., RT.
CONTROL SURFACE POSITIONS
T.E. DN, LT. DEG. T.E. UP, RT.
RUDDER PEDAL FORCE LBS
LAT & LONG. STICK FORCE LBS
RT. PULL
LT. PUSH
ALL Fs & FA DATA IN FRICTION BAND
LEFT
RIGHT
SIDESLIP FROM TRIM - DEG.

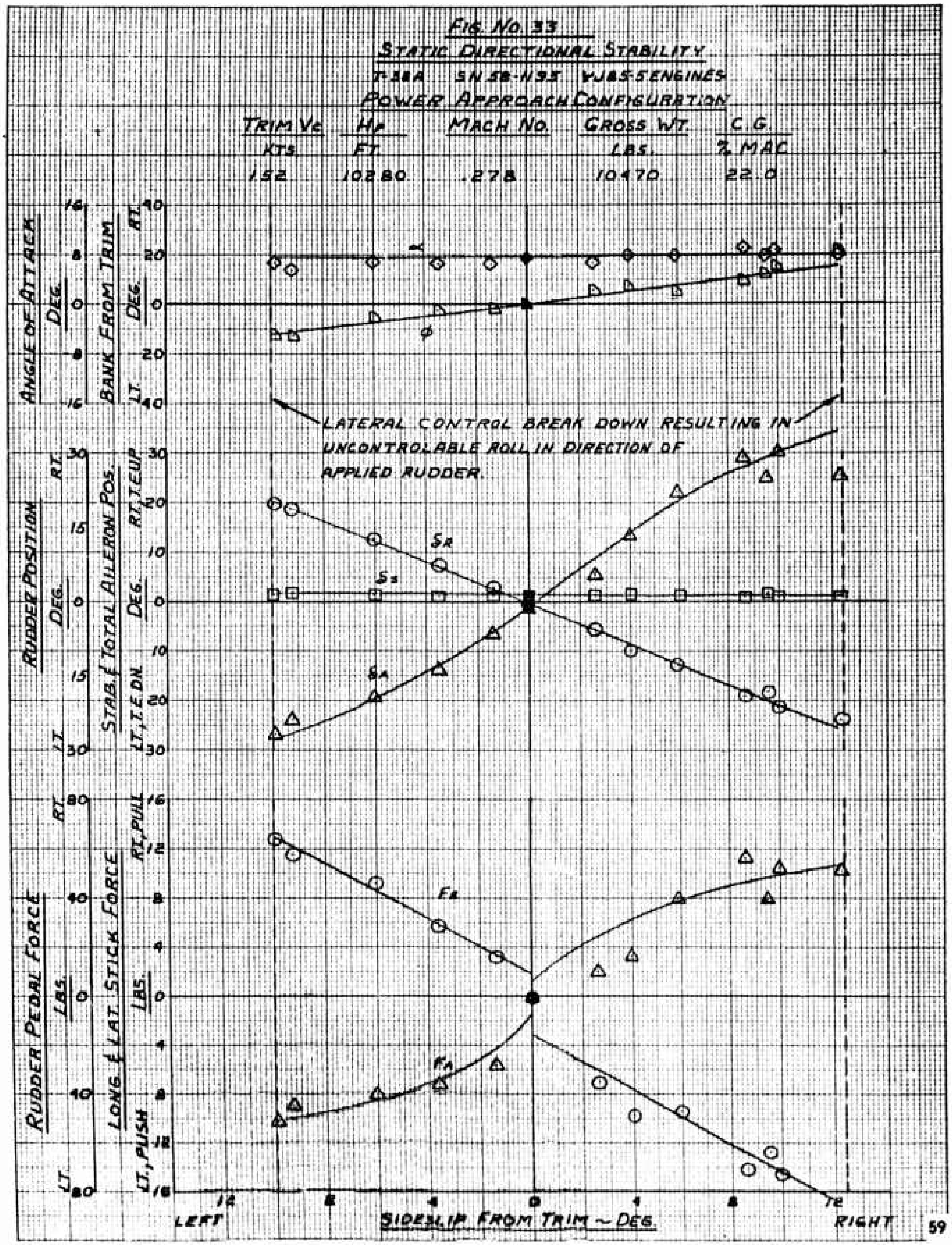


FIG. No. 33
STATIC DIRECTIONAL STABILITY
T-38A SN 58-1195 YJ85-5 ENGINES
POWER APPROACH CONFIGURATION
TRIM Vc KTS. 152
Hp FT. 10280
MACH NO. .278
GROSS WT. LBS. 10470
C.G. % MAC 22.0
ANGLE OF ATTACK DEG.
BANK FROM TRIM DEG. RT. LT.
LATERAL CONTROL BREAK DOWN RESULTING IN UNCONTROLABLE ROLL IN DIRECTION OF APPLIED RUDDER.
RUDDER POSITION DEG. RT. LT.
STAB. & TOTAL AILERON POS. DEG. RT., T.E. UP LT., T.E. DN
RUDDER PEDAL FORCE LBS. RT. LT.
LONG. & LAT. STICK FORCE LBS. RT., PULL LT., PUSH
SIDESLIP FROM TRIM ~ DEG.
LEFT
RIGHT

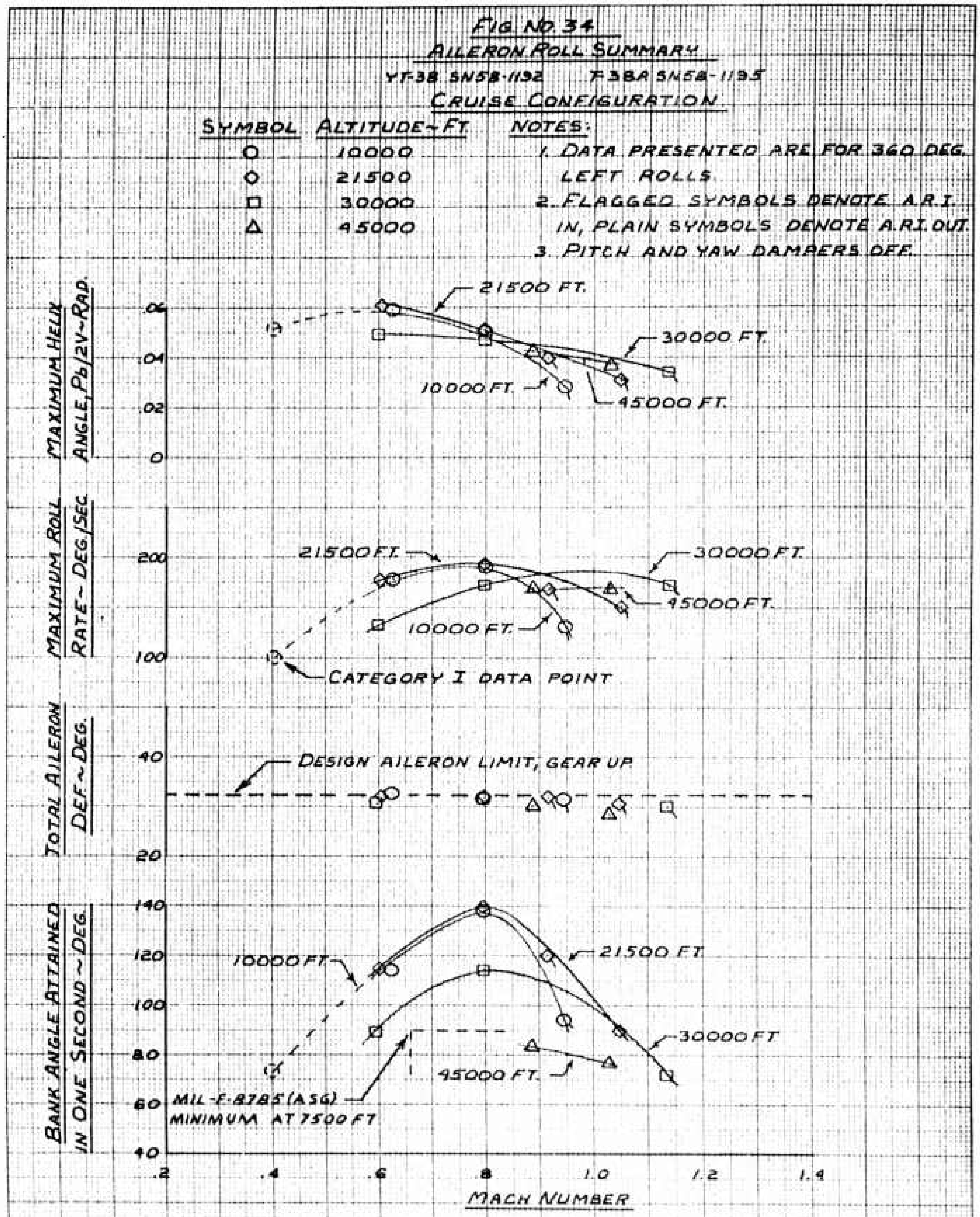


FIG. NO. 34
AILERON ROLL SUMMARY
YT-38 SN58-1192 T-38A SN58-1195
CRUISE CONFIGURATION
SYMBOL ALTITUDE ~ FT.
10000
21500
30000
45000
NOTES:
1. DATA PRESENTED ARE FOR 360 DEG. LEFT ROLLS
2. FLAGGED SYMBOLS DENOTE A.R.I. IN, PLAIN SYMBOLS DENOTE A.R.I. OUT.
3. PITCH AND YAW DAMPERS OFF.
MAXIMUM HELIX ANGLE, Pb/2V ~ RAD.
MAXIMUM ROLL RATE ~ DEG./SEC
TOTAL AILERON DEF. ~ DEG.
BANK ANGLE ATTAINED IN ONE SECOND ~ DEG.
21500 FT.
30000 FT.
10000 FT.
45000 FT.
CATEGORY I DATA POINT
DESIGN AILERON LIMIT, GEAR UP
MIL-F-8785 (ASG) MINIMUM AT 7500 FT
MACH NUMBER

Fig No. 35
Sideslip Angles During Rolling Maneuvers
YT-38 SN 58-1192 YJ85-1 Engines
Cruise Configuration
21500 Feet Altitude

Symbol	Aileron Rudder Interconnect	Pitch and Yaw Damper
Plain	Out	Off
Flagged	In	Off

Note: Data presented for 360 degree left rolls
Entry load factor ~ 1.9

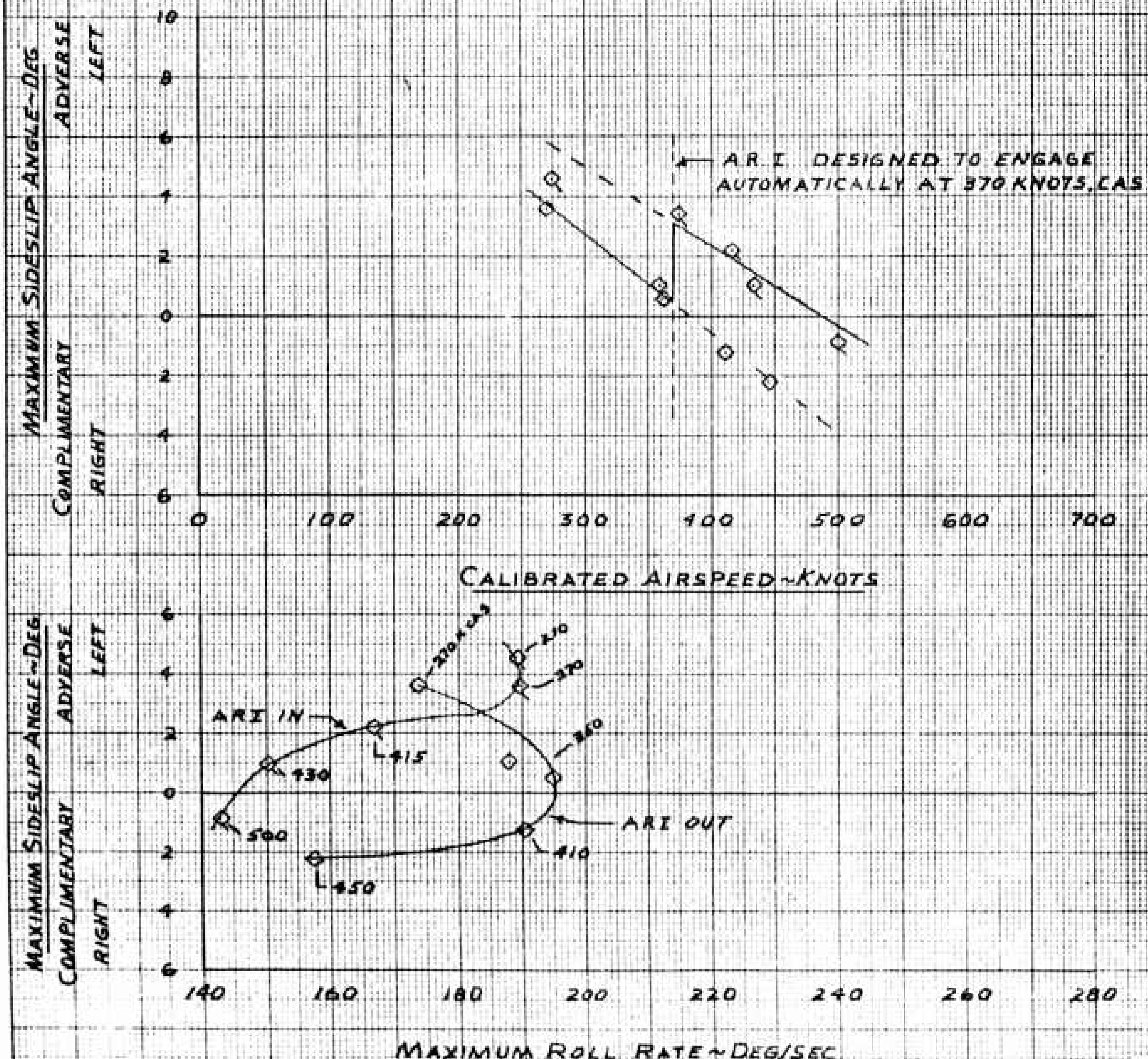


FIG. NO. 36

AILERON ROLL CHARACTERISTICS

XT-38 SN 58-1192 T-38A SN 58-1195

CRUISE CONFIGURATION

ALTITUDE ~ 21500 FEET

SYMBOL	ENTRY 'g'
□	0.0
○	1.0
△	3.0

NOTE:

1. DATA PRESENTED IS FOR FULL DEFLECTION LEFT ROLLS WITH DAMPERS OFF.
2. FLAGGED SYMBOLS DENOTE A.R.I. IN.
3. PLAIN SYMBOLS DENOTE A.R.I. OUT.
4. SOLID LINES REPRESENT FAIRING FOR A.R.I. IN NORMAL POSITION.

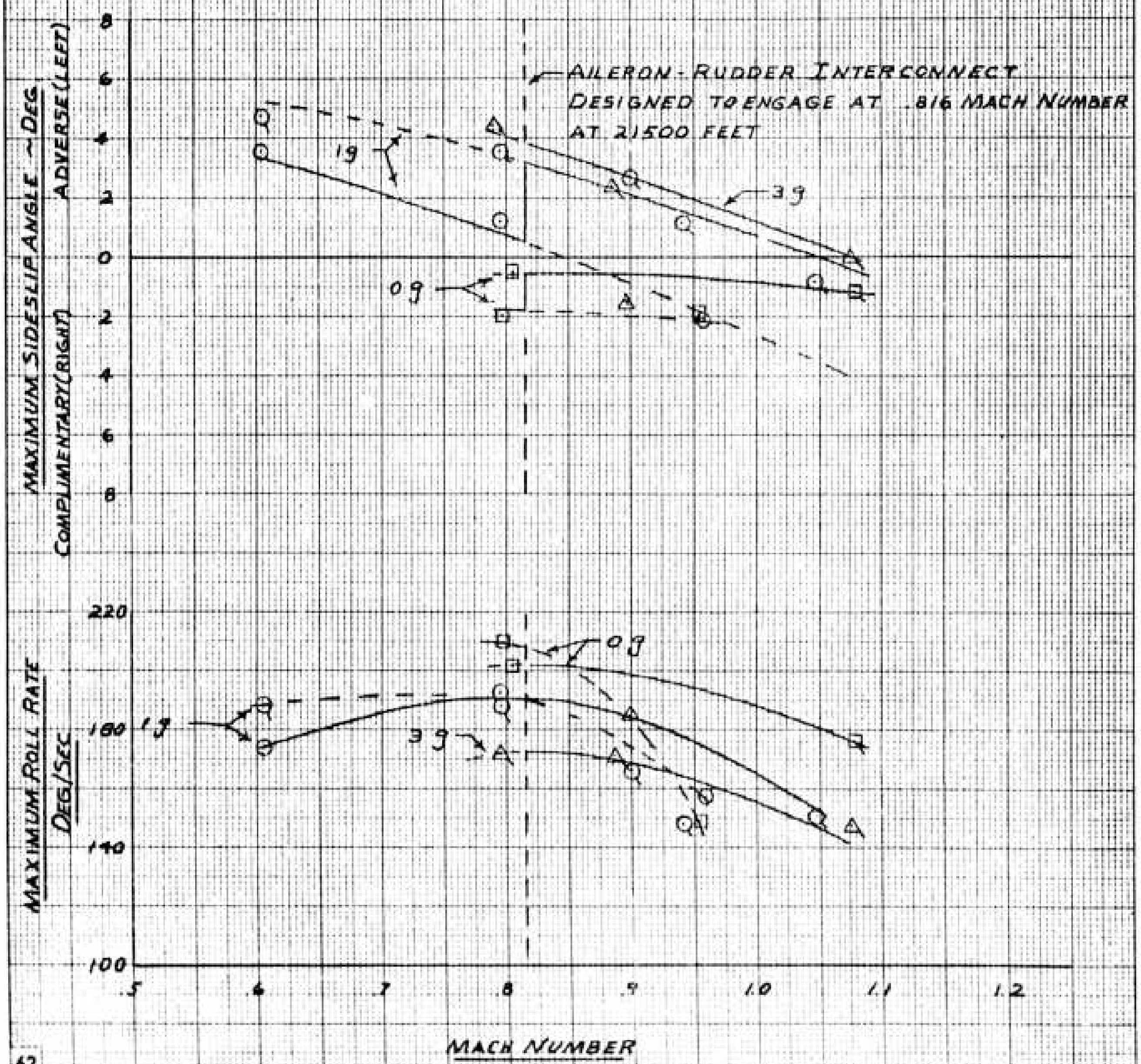


FIG. NO. 37
AILERON ROLL CHARACTERISTICS
T-38A SN 58-1195 YJ85-5 ENGINES
TAKE-OFF CONFIGURATION

SYMBOL	TRIM Vc KTS	HP FT.	MACH NO.	ENTRY 'g'	GROSS WT. LBS	C.G. % MAC
○	151	10420	.279	1.0	9360	23.5
□	149	10260	.272	1.0	9380	23.7

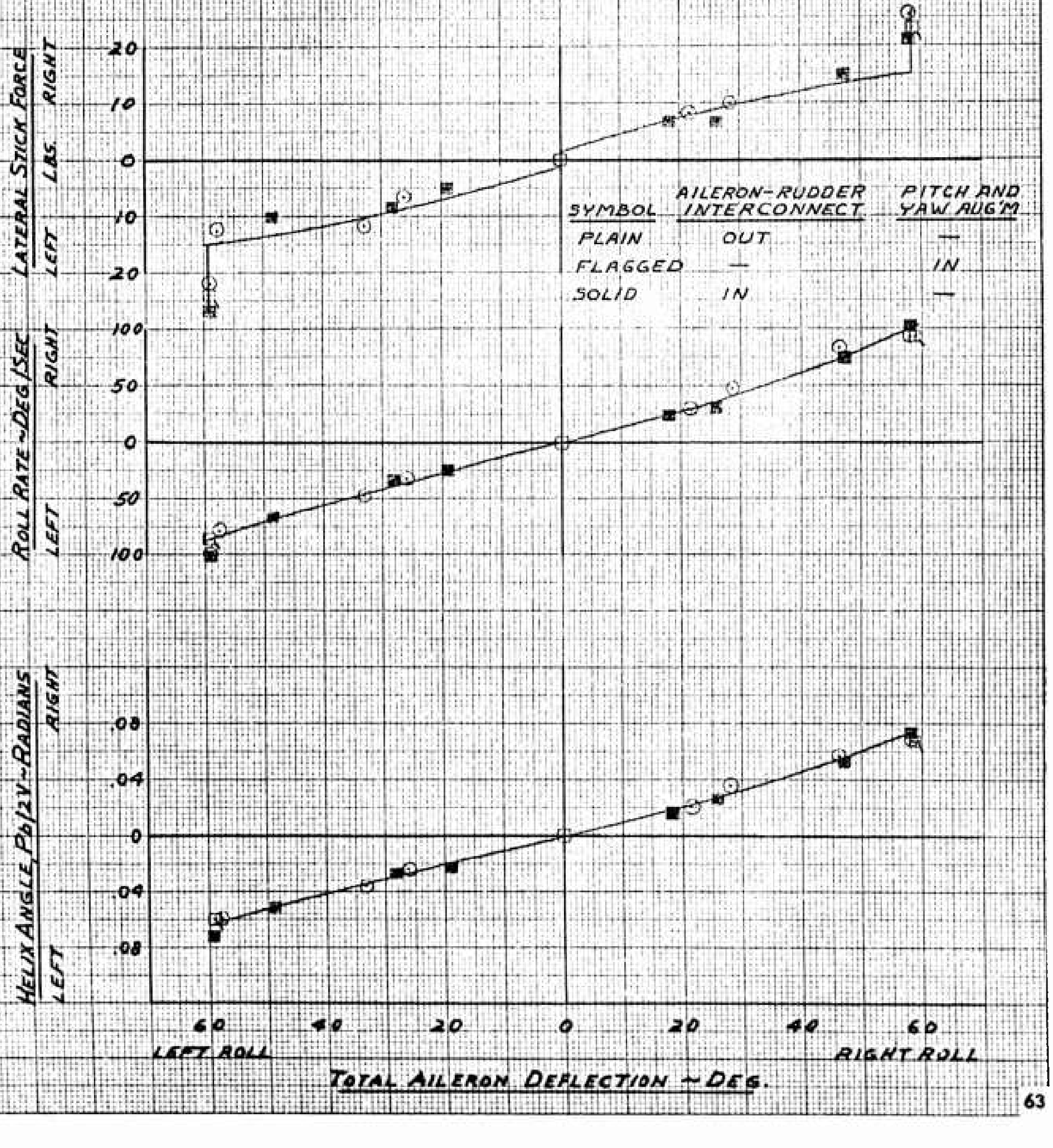
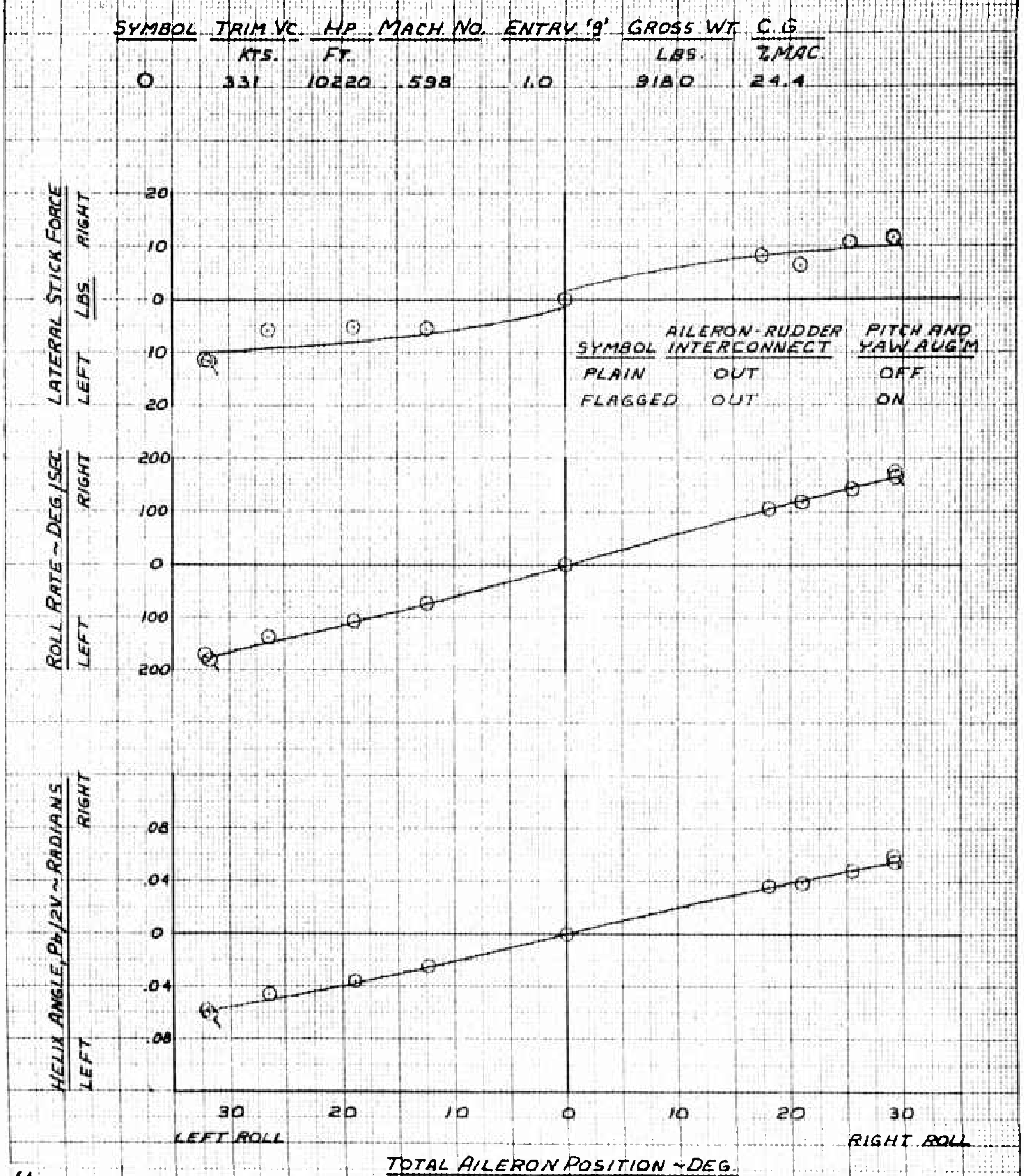


FIG. NO. 38

AILERON ROLL CHARACTERISTICS

YT-38 SN58-1192 YJ85-1 ENGINES

CRUISE CONFIGURATION



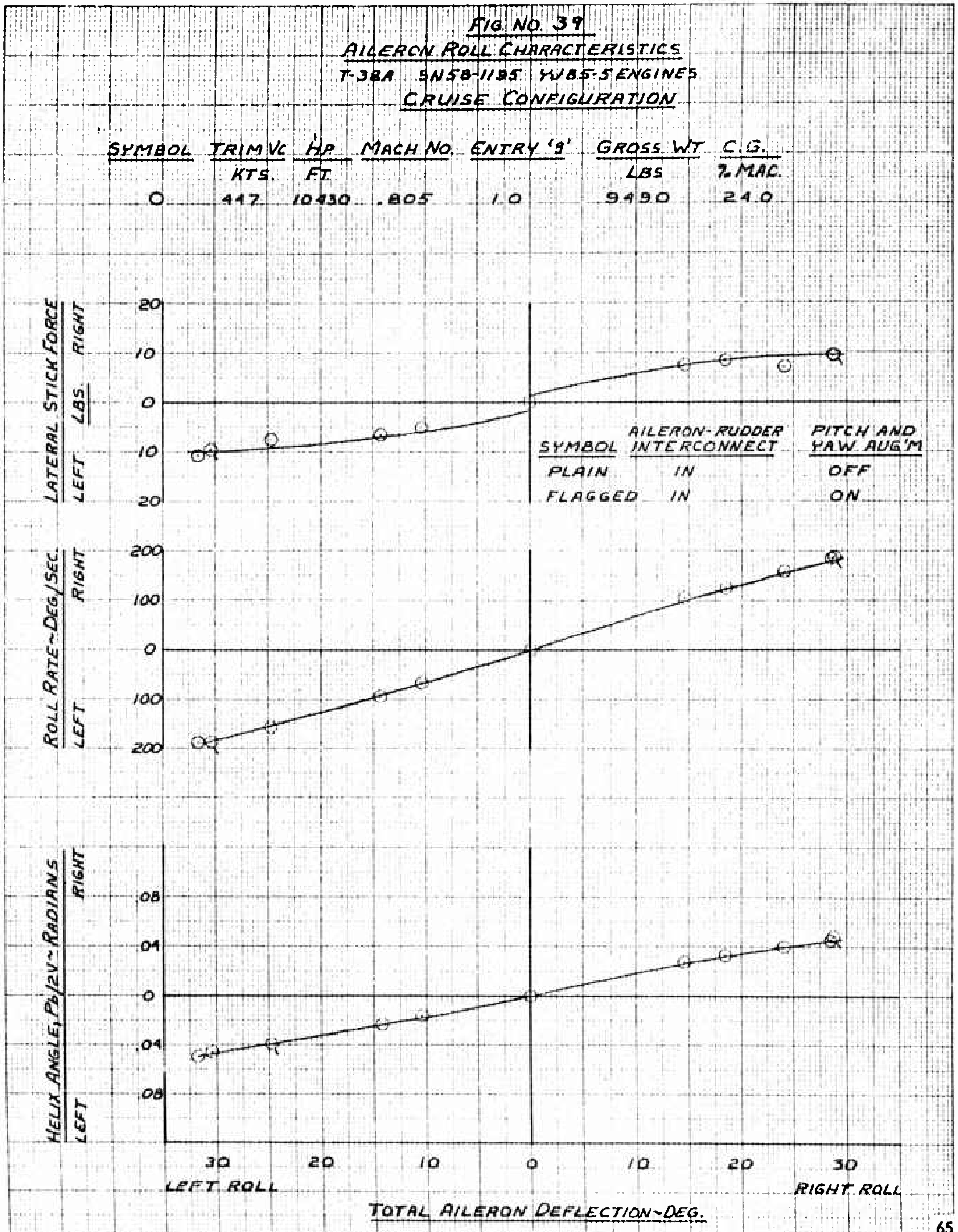


FIG. NO. 39
AILERON ROLL CHARACTERISTICS
T-38A SN58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION
SYMBOL TRIM Vc KTS. Hp FT. MACH NO. ENTRY 'g' GROSS WT LBS C.G. % MAC.
O 447 10430 .805 1.0 9490 24.0
LATERAL STICK FORCE LEFT LBS. RIGHT
20
10
0
10
20
SYMBOL
AILERON-RUDDER INTERCONNECT
PITCH AND YAW AUG'M
PLAIN IN OFF
FLAGGED IN ON
ROLL RATE~DEG/SEC LEFT RIGHT
200
100
0
100
200
HELIX ANGLE, Pb/2V~RADIANS LEFT RIGHT
.08
.04
0
.04
.08
30
20
10
0
10
20
30
LEFT ROLL
RIGHT ROLL
TOTAL AILERON DEFLECTION~DEG.

FIG. NO. 40

AILERON ROLL CHARACTERISTICS

T-38A SN58-1195 YJ85-5 ENGINES

CRUISE CONFIGURATION

SYMBOL	TRIM Vc KTS	HP FT.	MACH NO.	ENTRY 'g'	GROSS WT LBS.	C. G. % MAC.
O	533	10290	.949	1.0	11330	15.6

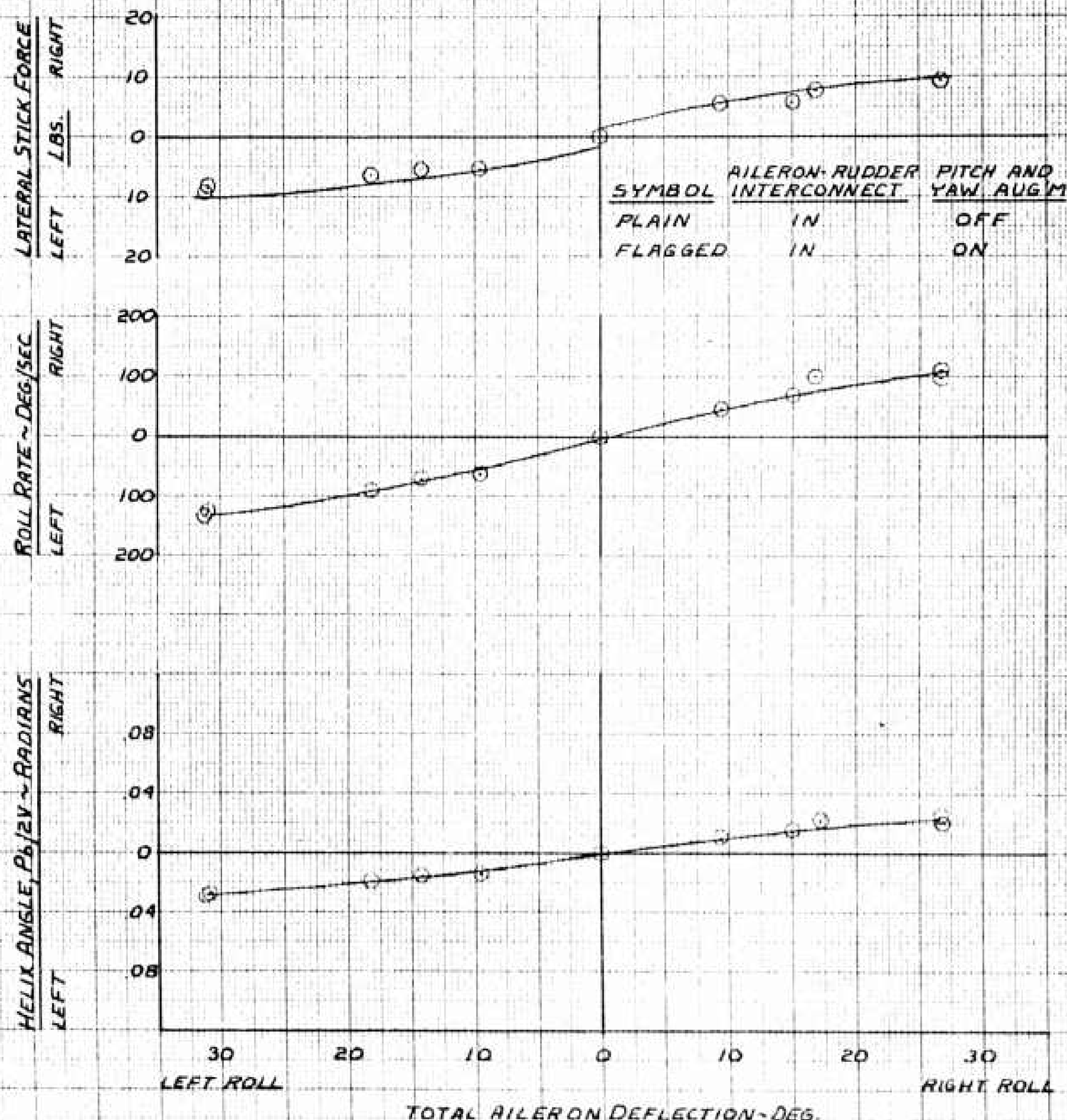


FIG. NO. 41

AILERON ROLL CHARACTERISTICS

YT-38 S/N 58-1192 YJ85-1 ENGINES

CRUISE CONFIGURATION

SYMBOL	TRIM V_c KTS	HP FT.	MACH NO.	ENTRY 'g'	GROSS WT. LBS.	C.G. % MAC.
□	359	21820	.793	0.0	10600	14.8
○	358	21850	.795	1.0	9590	15.6
△	357	21400	.787	3.0	10690	14.7

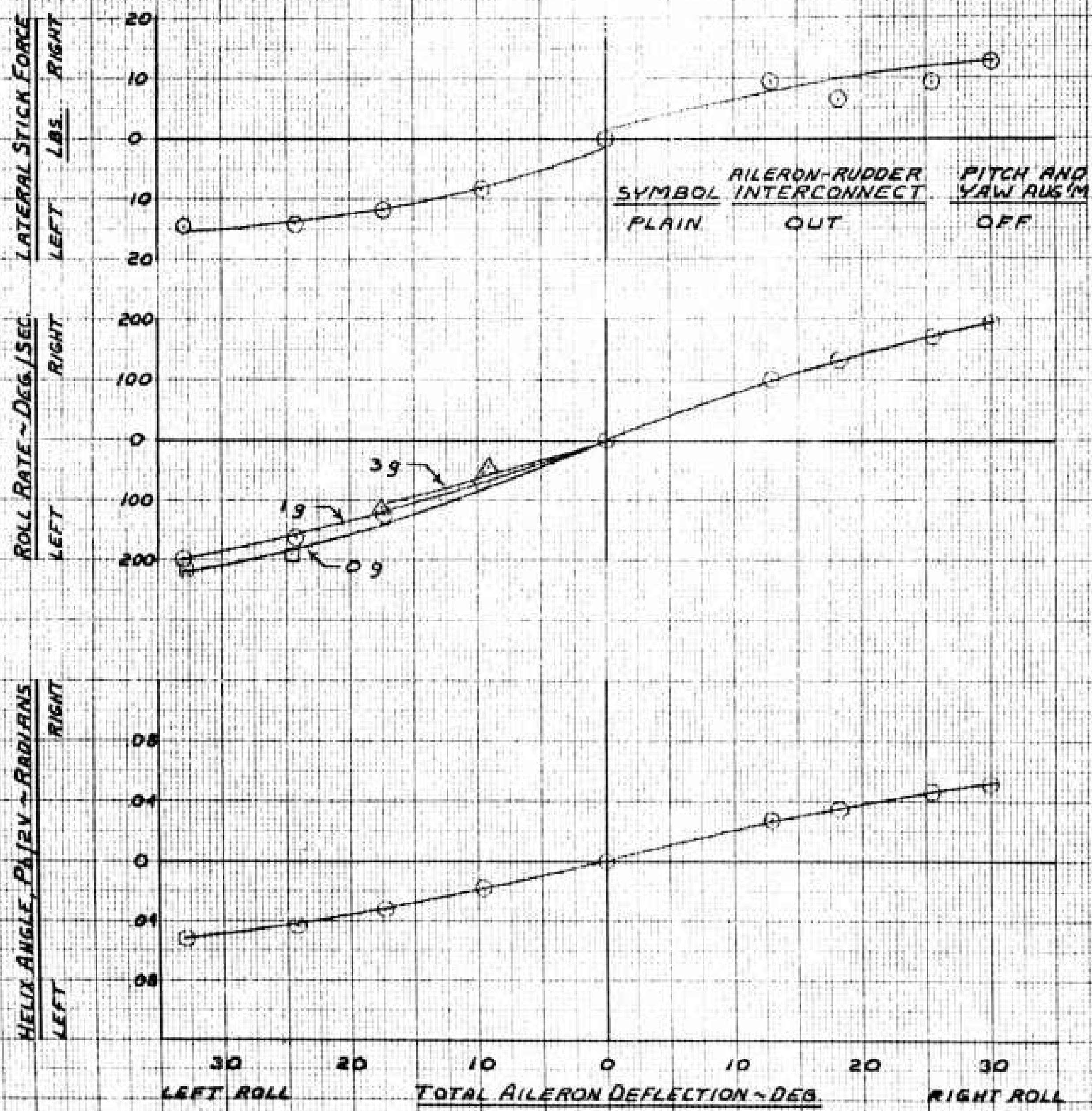


FIG. NO. 42

AILERON ROLL CHARACTERISTICS

YT-38 SN 58-1192 YJ85-1 ENGINES

CRUISE CONFIGURATION

SYMBOL	TRIM V_c KTS	HP FT	MACH NO	ENTRY 'g'	GROSS WT LBS.	C.G. % MAC
□	410	21860	.900	0.0	10370	15.1
○	407	22070	.900	1.0	9820	15.3
△	407	22130	.901	3.0	10270	15.2

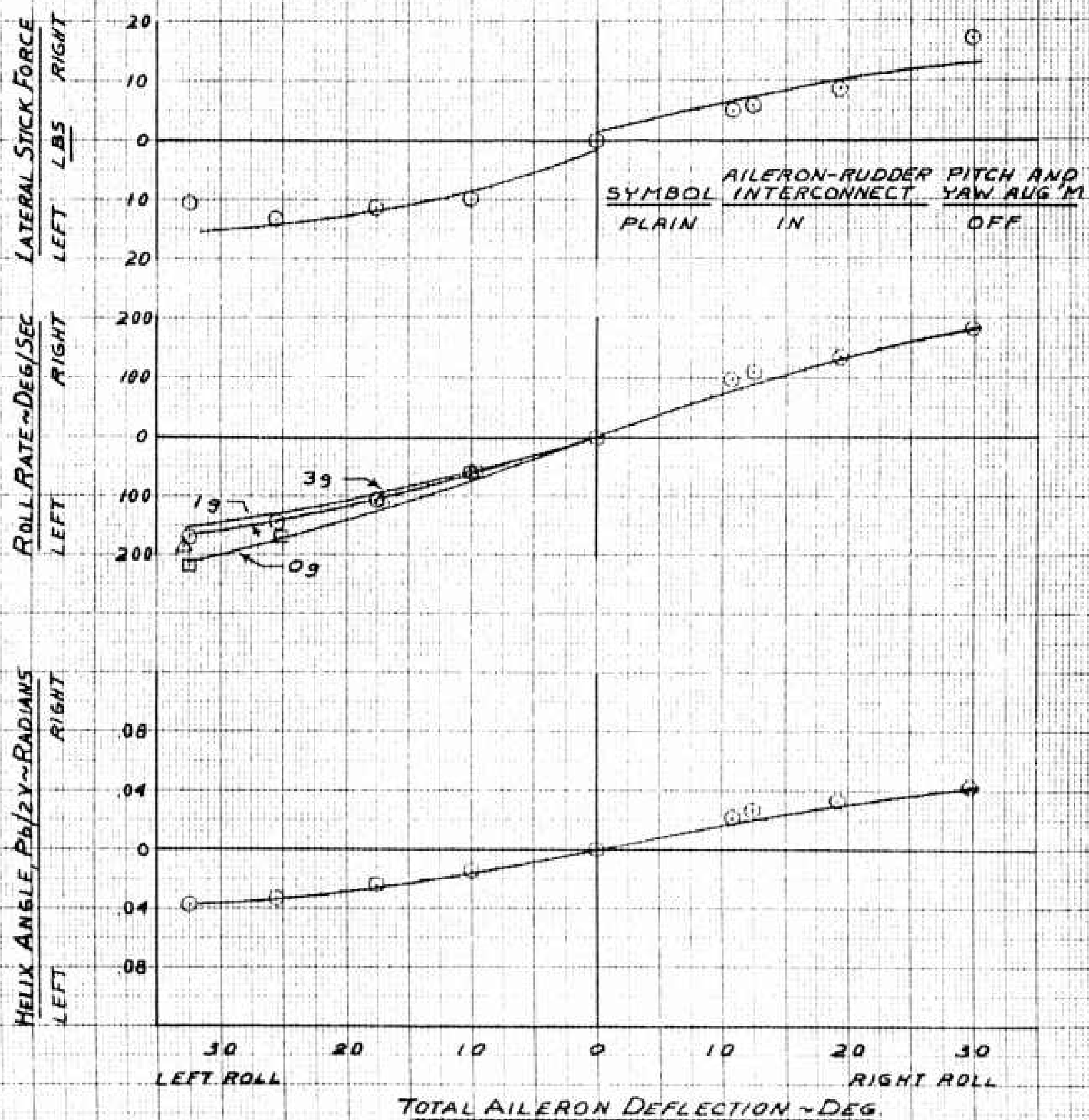


Fig. No. 43
AILERON ROLL CHARACTERISTICS
T-38R SN58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION

SYMBOL	TRIM Vc KTS.	Hp FT.	MACH NO.	ENTRY 'g'	GROSS WT. LBS.	C.G. % MAC.
O	499	21240	1.068	1.0	10780	22.1

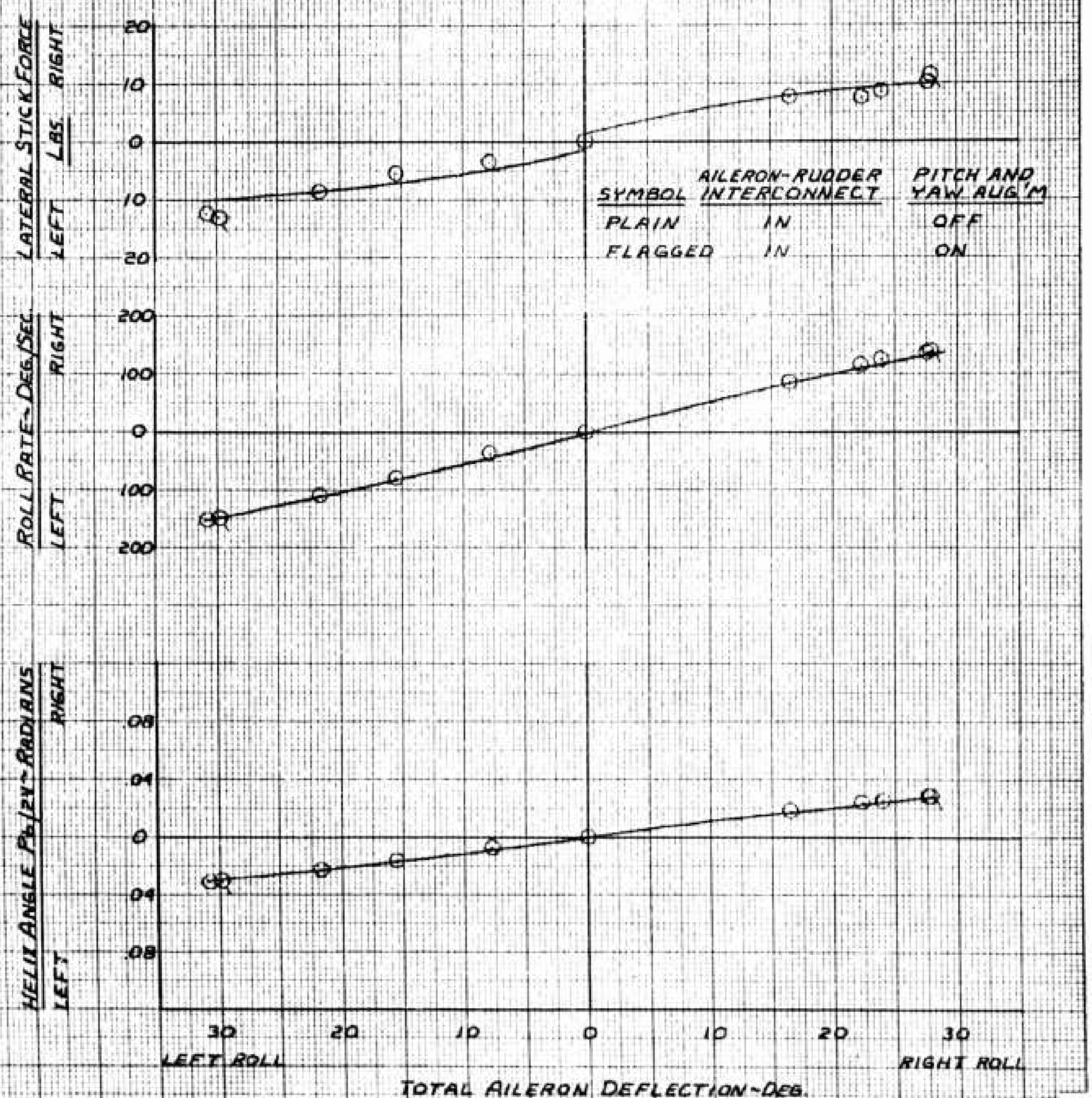


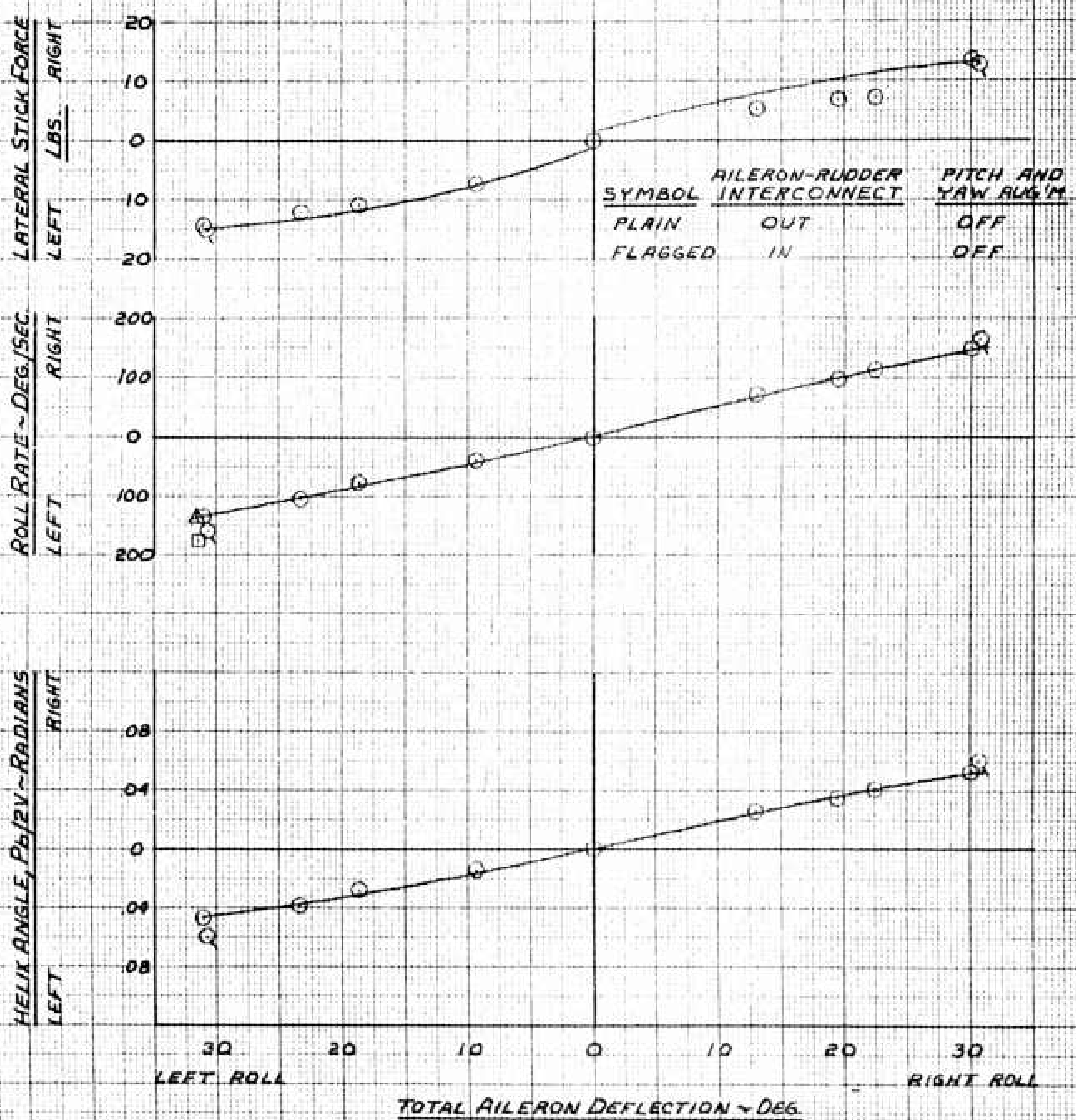
FIG. NO. 44

AILERON ROLL CHARACTERISTICS

YT-38 SN58-1192 YJ85-1 ENGINES

CRUISE CONFIGURATION

SYMBOL	TRIM Vc KTS.	HP FT.	MACH NO.	ENTRY 'g'	GROSS WT. LBS.	C.G. % MAC
□	230	30210	.623	0.0	9970	15.4
○	223	30990	.611	1.0	10450	14.9
△	223	30130	.607	1.6	9710	15.3



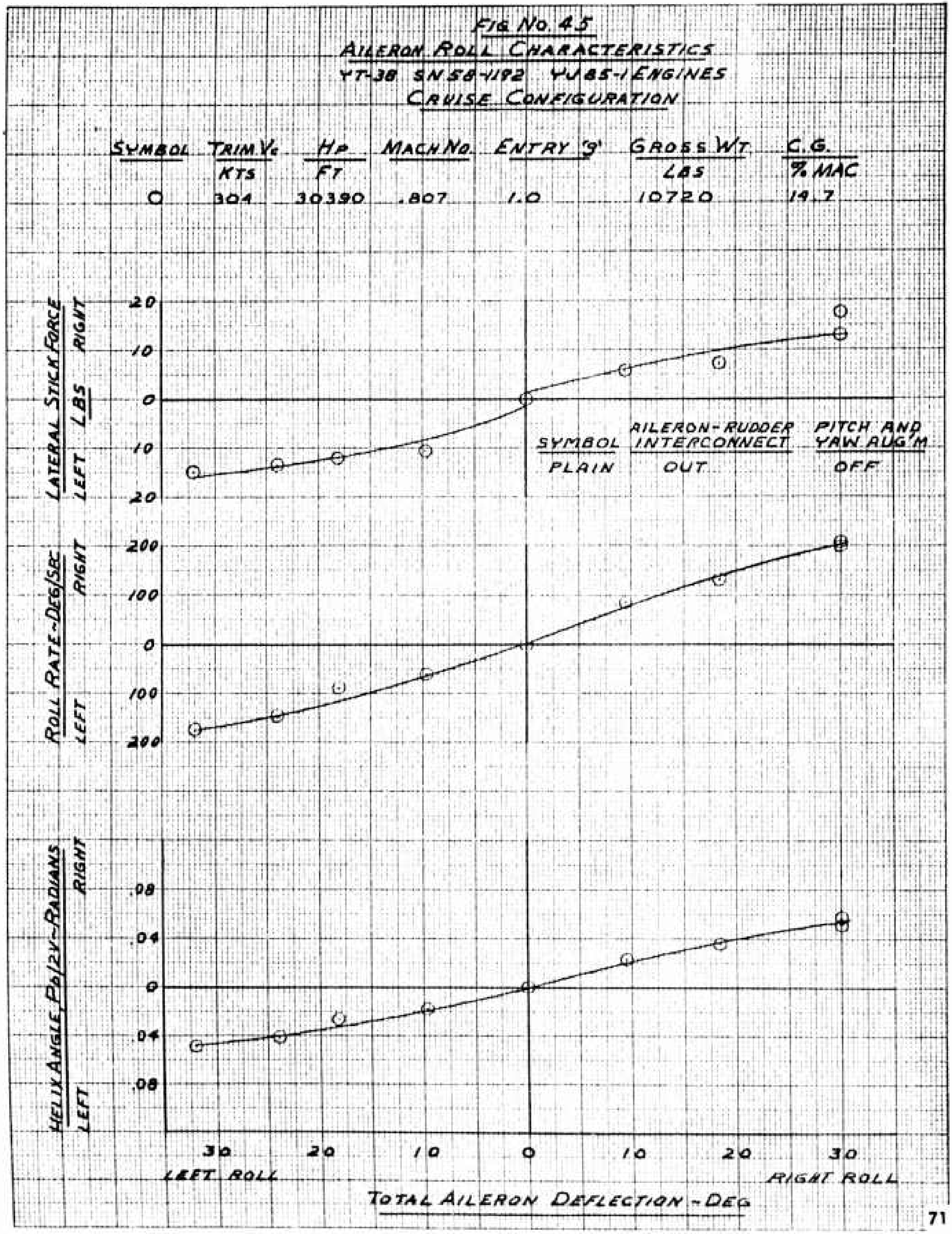


FIG No. 45
AILERON ROLL CHARACTERISTICS
YT-38 SN 58-1192 YJ85-1 ENGINES
CRUISE CONFIGURATION
SYMBOL
TRIM V_c KTS
H_P FT
MACH NO.
ENTRY 'g'
GROSS WT LBS
C.G. % MAC
304
30390
.807
1.0
10720
14.7
LATERAL STICK FORCE LBS
LEFT
RIGHT
ROLL RATE - DEG/SEC
HELIX ANGLE Pb/2V - RADIANS
SYMBOL
AILERON-RUDDER INTERCONNECT
PITCH AND YAW AUG'M
PLAIN
OUT
OFF
LEFT ROLL
RIGHT ROLL
TOTAL AILERON DEFLECTION - DEG

FIG. NO. 46

AILERON ROLL CHARACTERISTICS

T-38A SN 58-1195 YJ85-5 ENGINES

CRUISE CONFIGURATION

SYMBOL	TRIM Vc KTS.	HP FT.	MACH NO.	ENTRY 'g'	GROSS WT. LBS.	C.G. % MAC.
O	450	30130	1.138	1.0	10800	14.9

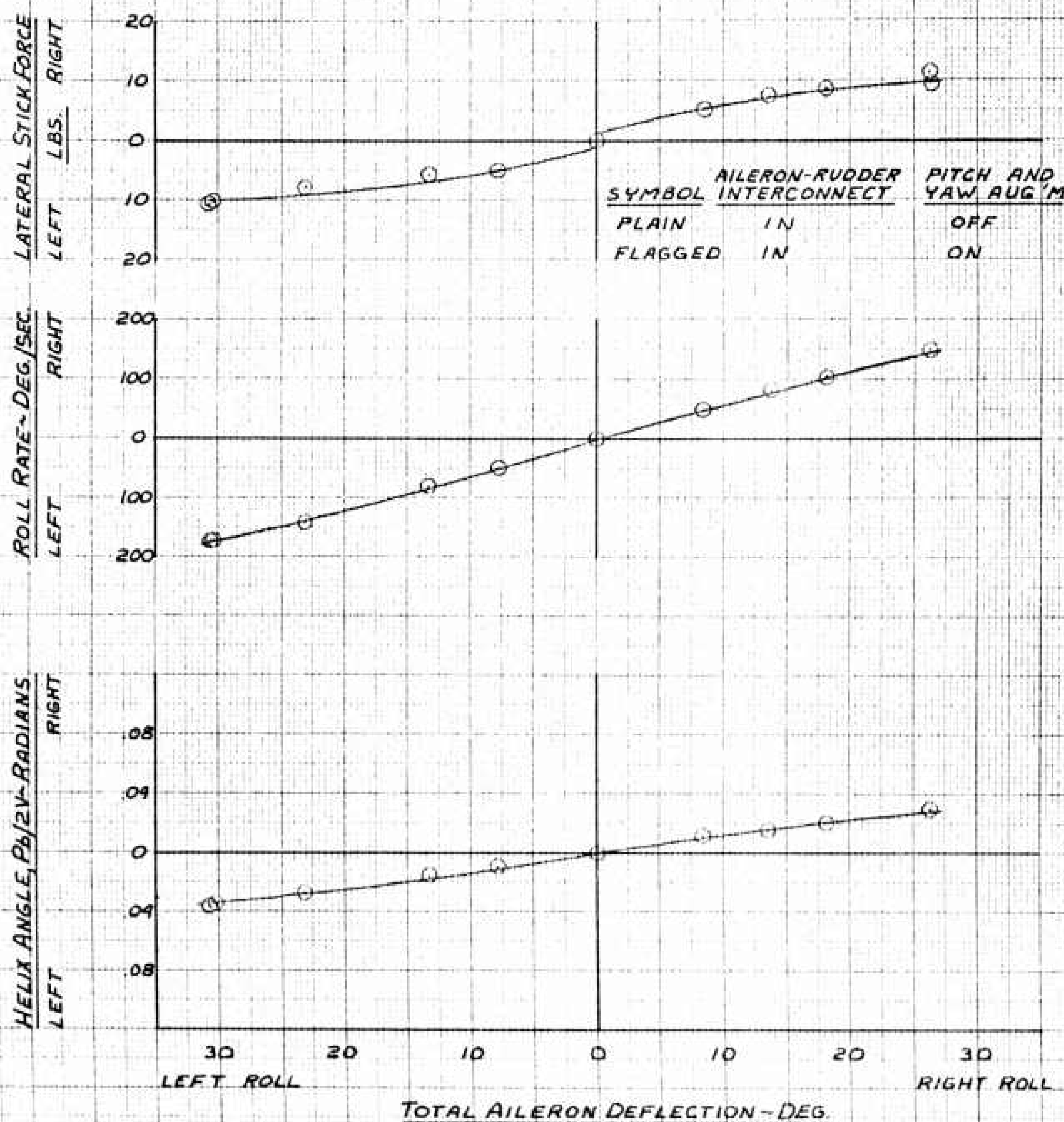


FIG. NO. 47
AILERON ROLL CHARACTERISTICS
T-38A SN58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION

SYMBOL	TRIM Vc KTS.	Hp FT.	MACH NO.	ENTRY 'g'	GROSS WT.	C.G. % MAC.
○	242	45470	.894	1.0	9410	15.7

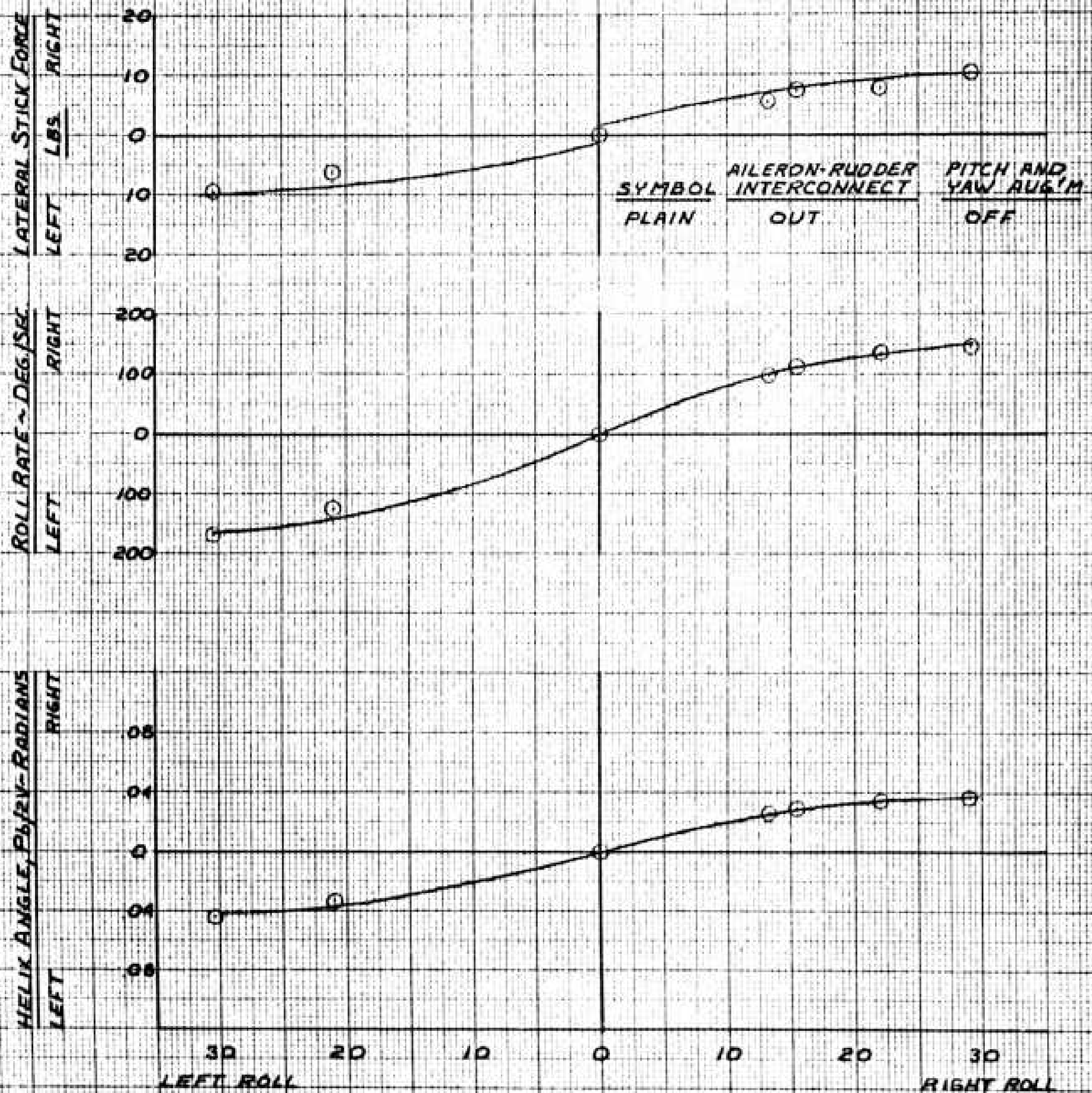


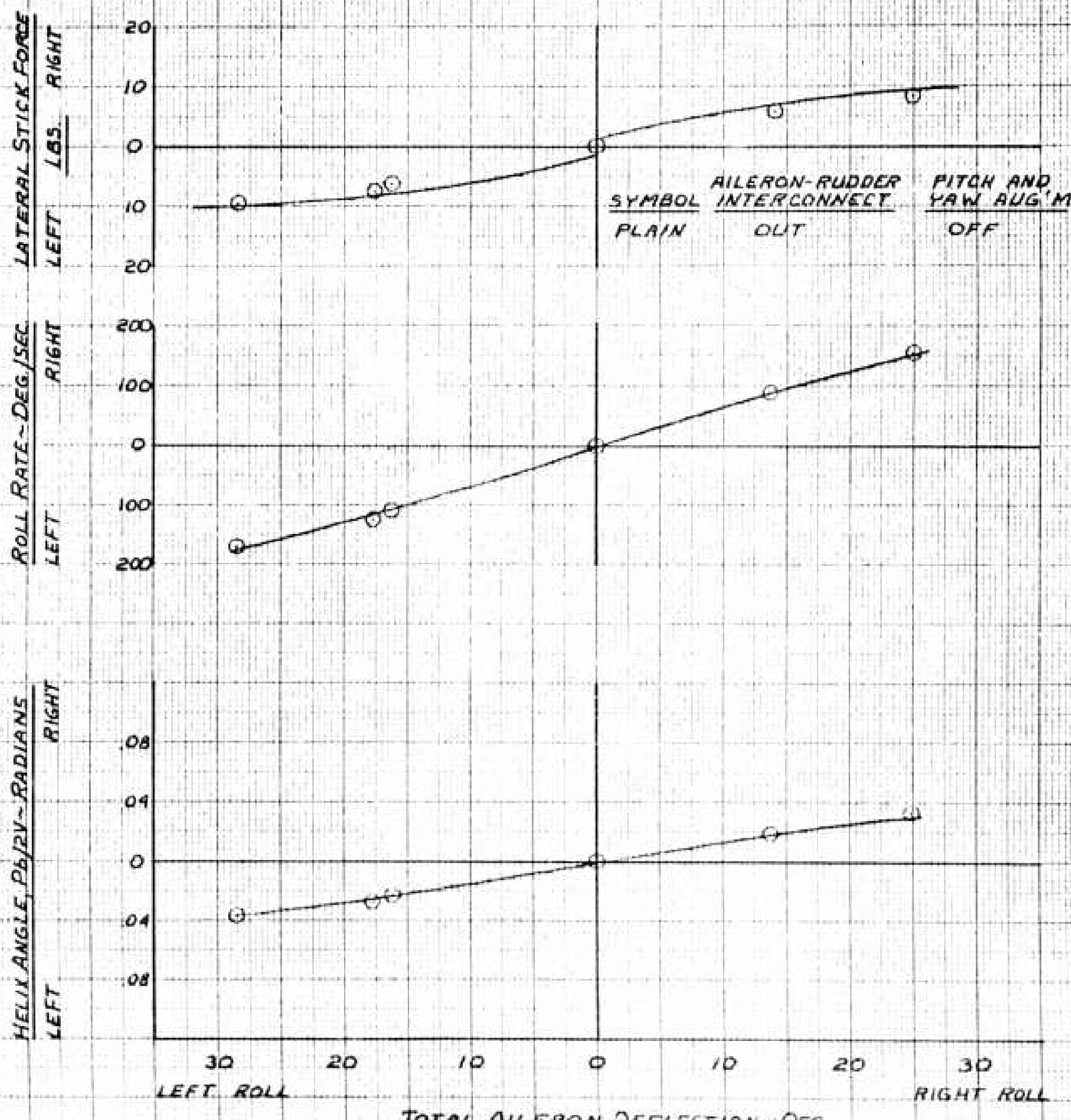
FIG. NO. 48

AILERON ROLL CHARACTERISTICS

T-38A SN58-1195 YJ85-5 ENGINES

CRUISE CONFIGURATION

SYMBOL	TRIM Vc KTS.	HP FT.	MACH NO.	ENTRY 'g'	GROSS WT LBS.	C.G. % MAC.
O	304	44970	1.073	1.0	10760	14.9



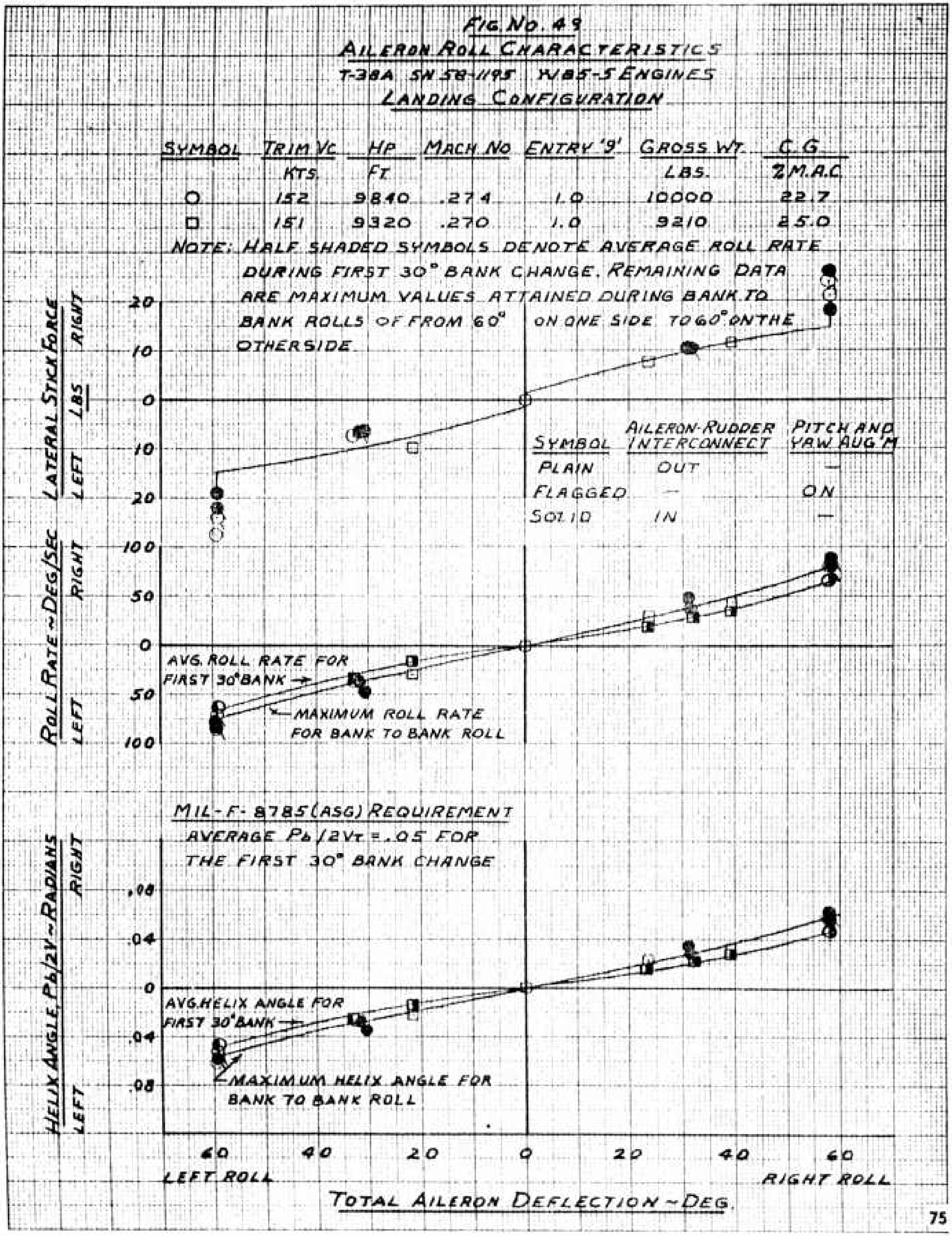


FIG. NO. 49
AILERON ROLL CHARACTERISTICS
T-38A SN 58-1195 YJ85-5 ENGINES
LANDING CONFIGURATION
SYMBOL TRIM Vc KTS. HP FT. MACH NO ENTRY 'g' GROSS WT. LBS. C.G %M.A.C.
○ 152 9840 .274 1.0 10000 22.7
□ 151 9320 .270 1.0 9210 25.0
NOTE: HALF SHADED SYMBOLS DENOTE AVERAGE ROLL RATE DURING FIRST 30° BANK CHANGE. REMAINING DATA ARE MAXIMUM VALUES ATTAINED DURING BANK TO BANK ROLLS OF FROM 60° ON ONE SIDE TO 60° ON THE OTHER SIDE.
SYMBOL AILERON-RUDDER INTERCONNECT PITCH AND YAW AUG'M
PLAIN OUT —
FLAGGED — ON
SOLID IN —
LATERAL STICK FORCE LBS
LEFT RIGHT
20
10
0
10
20
ROLL RATE ~ DEG/SEC
LEFT RIGHT
100
50
0
50
100
AVG. ROLL RATE FOR FIRST 30° BANK
MAXIMUM ROLL RATE FOR BANK TO BANK ROLL
MIL-F-8785(ASG) REQUIREMENT
AVERAGE Pb/2VT = .05 FOR THE FIRST 30° BANK CHANGE
HELIX ANGLE, Pb/2V ~ RADIANS
LEFT RIGHT
.08
.04
0
.04
.08
AVG. HELIX ANGLE FOR FIRST 30° BANK
MAXIMUM HELIX ANGLE FOR BANK TO BANK ROLL
60 40 20 0 20 40 60
LEFT ROLL
RIGHT ROLL
TOTAL AILERON DEFLECTION ~ DEG.

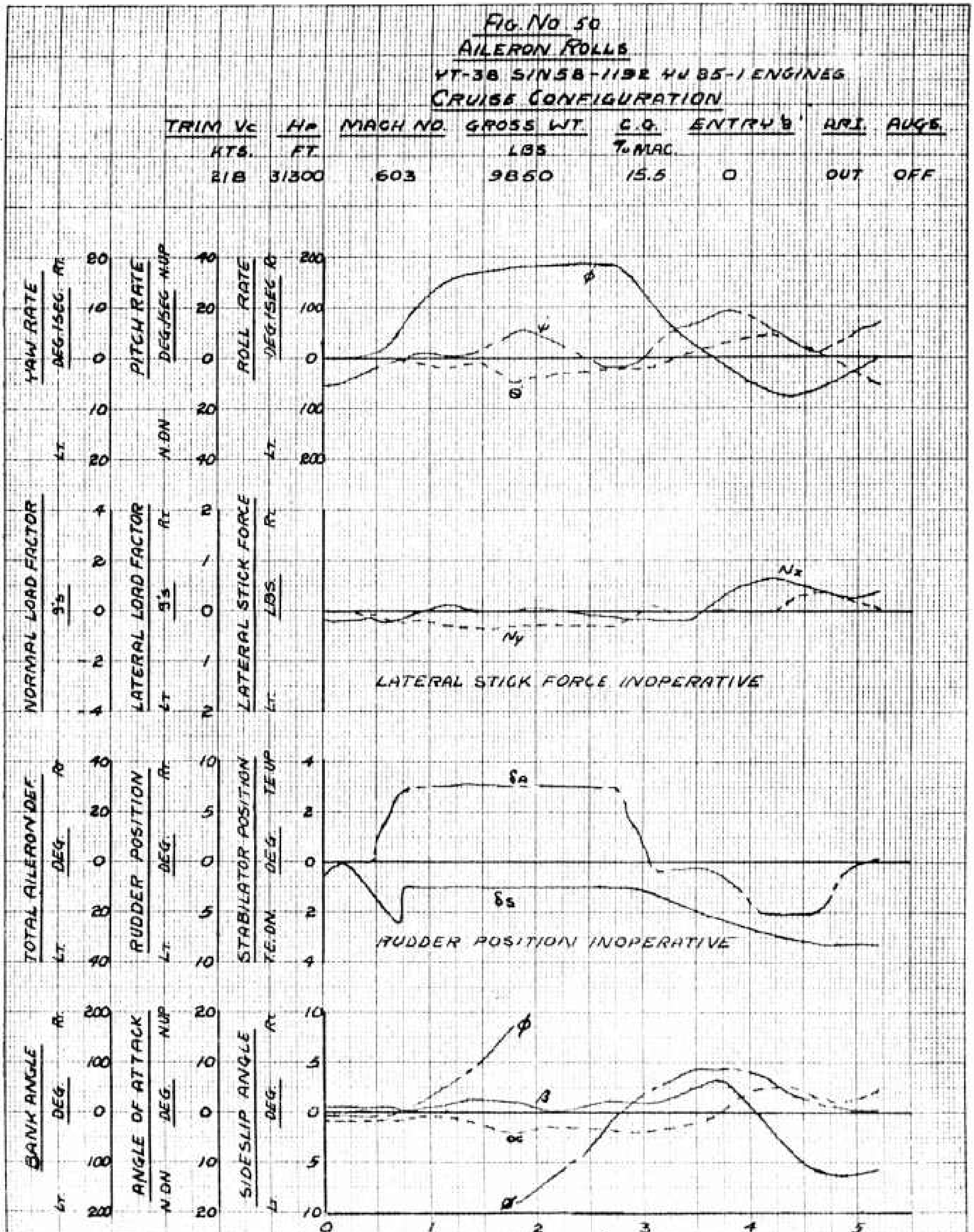


FIG. NO. 50
AILERON ROLLS
YT-38 S/N 58-1192 YJ 85-1 ENGINES
CRUISE CONFIGURATION
TRIM Vc KTS. 218
Hp FT. 31300
MACH NO. .603
GROSS WT. LBS. 9850
C.G. % MAC 15.5
ENTRY g's 0
ARI. OUT
AUGS. OFF
YAW RATE DEG/SEC. RT. LT.
PITCH RATE DEG/SEC N.UP N.DN
ROLL RATE DEG/SEC RT. LT.
NORMAL LOAD FACTOR g's
LATERAL LOAD FACTOR g's RT. LT.
LATERAL STICK FORCE LBS. RT. LT.
LATERAL STICK FORCE INOPERATIVE
TOTAL AILERON DEF. DEG. RT. LT.
RUDDER POSITION DEG. RT. LT.
STABILATOR POSITION DEG. T.E.UP T.E.DN.
RUDDER POSITION INOPERATIVE
BANK ANGLE DEG. RT. LT.
ANGLE OF ATTACK DEG. N.UP N.DN
SIDESLIP ANGLE DEG. RT. LT.
TIME ~ SECONDS

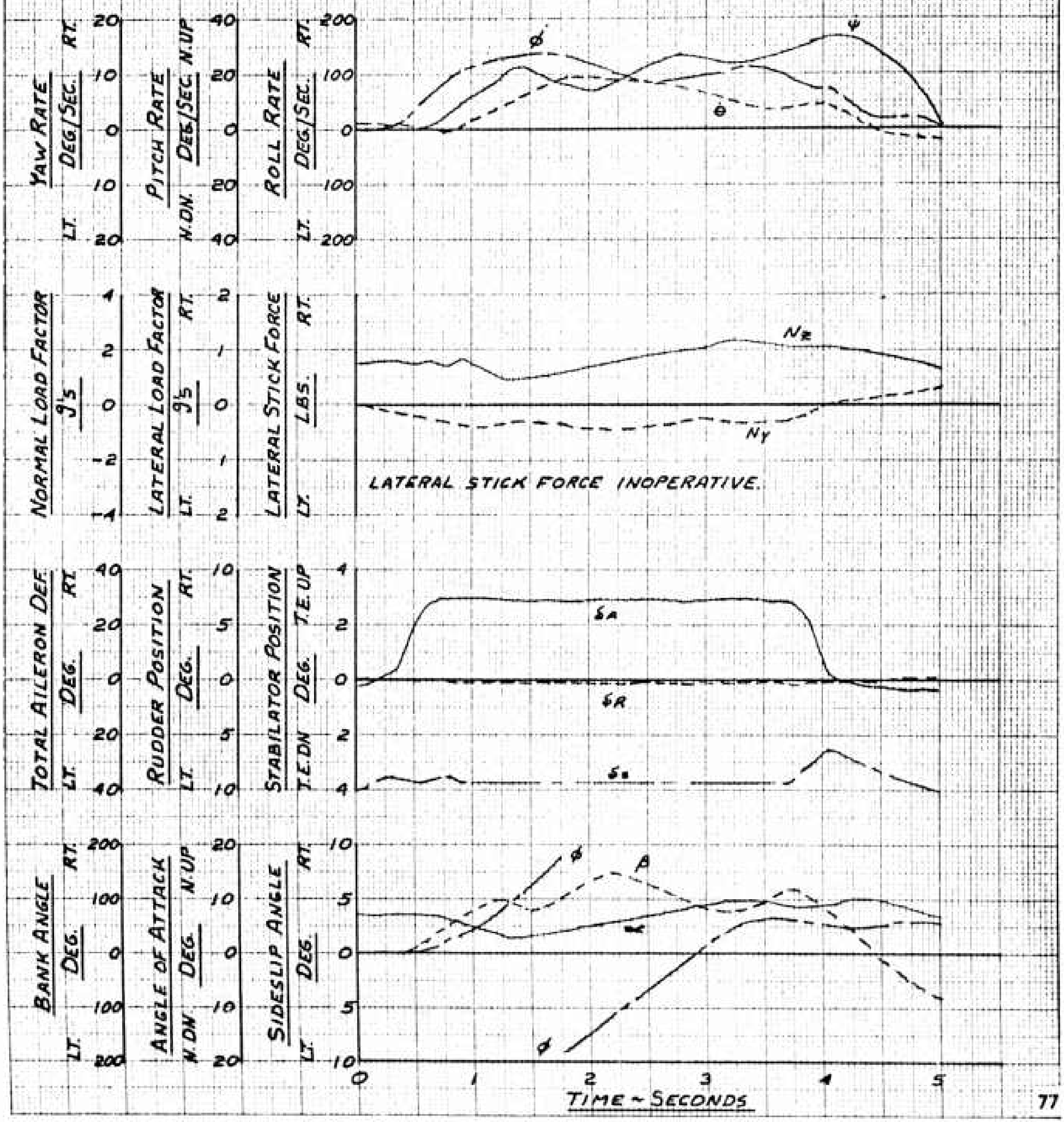
FIG. NO. 51

AILERON ROLLS

YT-38 SN58-1192 YJ85-1 ENGINES

CRUISE CONFIGURATION

TRIM VC KTS.	HP FT.	MACH NO.	GROSS WT. LBS.	C.G. % M.A.C.	ENTRY 'g'	ARI.	AUGS.
224	29980	.601	9410	15.6	1.6	OUT	OFF



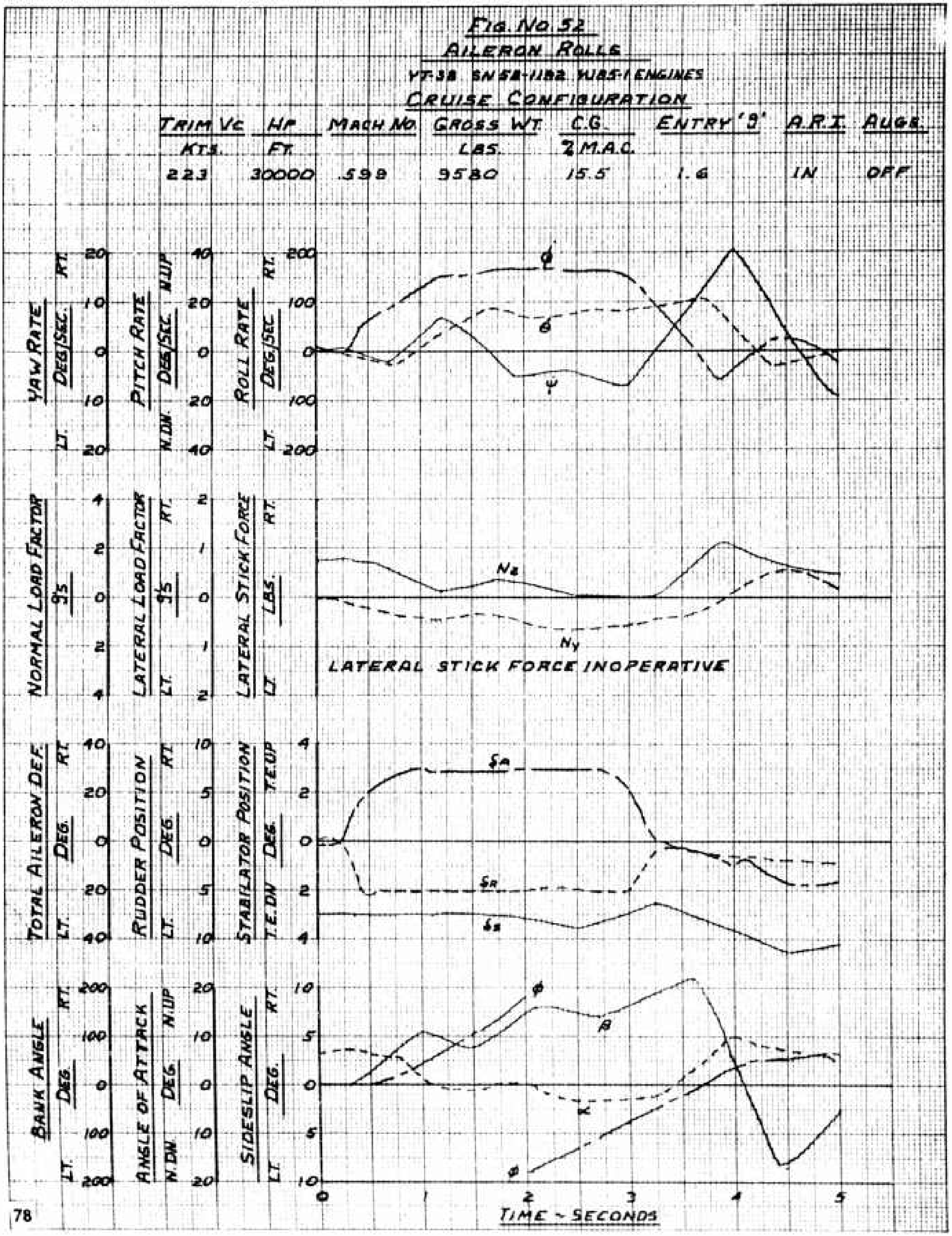


FIG. NO. 52
AILERON ROLLS
YT-38 SN 58-1182 YJ85-1 ENGINES
CRUISE CONFIGURATION
TRIM VC KTS. 223
HP FT. 30000
MACH NO. .598
GROSS WT LBS. 9580
C.G. % M.A.C. 15.5
ENTRY 'g' 1.6
A.R.I. IN
AUGS. OFF
YAW RATE DEG/SEC. LT. RT.
PITCH RATE DEG/SEC. N.DN. N.UP
ROLL RATE DEG/SEC. LT. RT.
NORMAL LOAD FACTOR g's
LATERAL LOAD FACTOR g's LT. RT.
LATERAL STICK FORCE LBS. LT. RT.
LATERAL STICK FORCE INOPERATIVE
TOTAL AILERON DEF. DEG. LT. RT.
RUDDER POSITION DEG. LT. RT.
STABILATOR POSITION DEG. T.E.DN T.E.UP
BANK ANGLE DEG. LT. RT.
ANGLE OF ATTACK DEG. N.DN. N.UP
SIDESLIP ANGLE DEG. LT. RT.
TIME ~ SECONDS

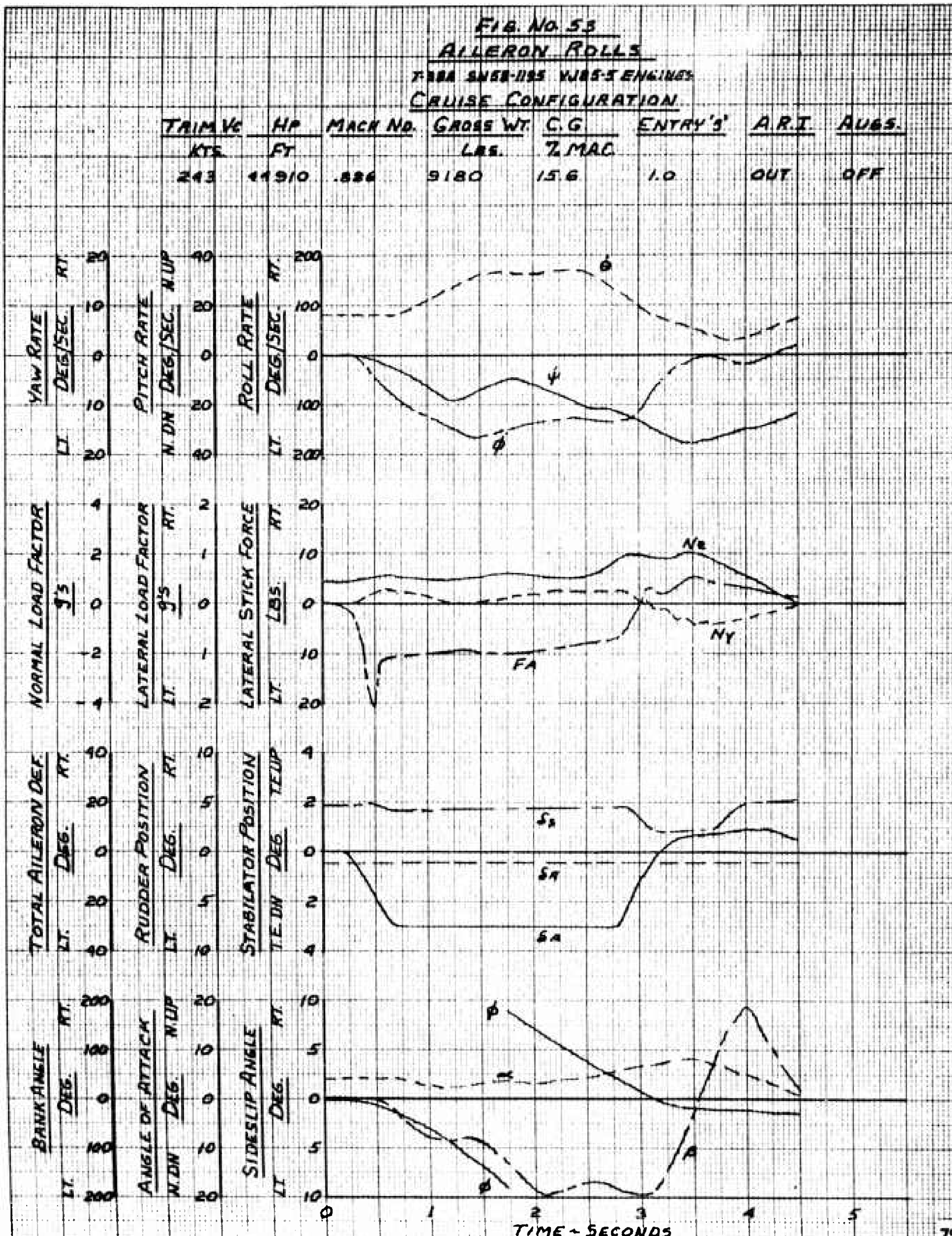
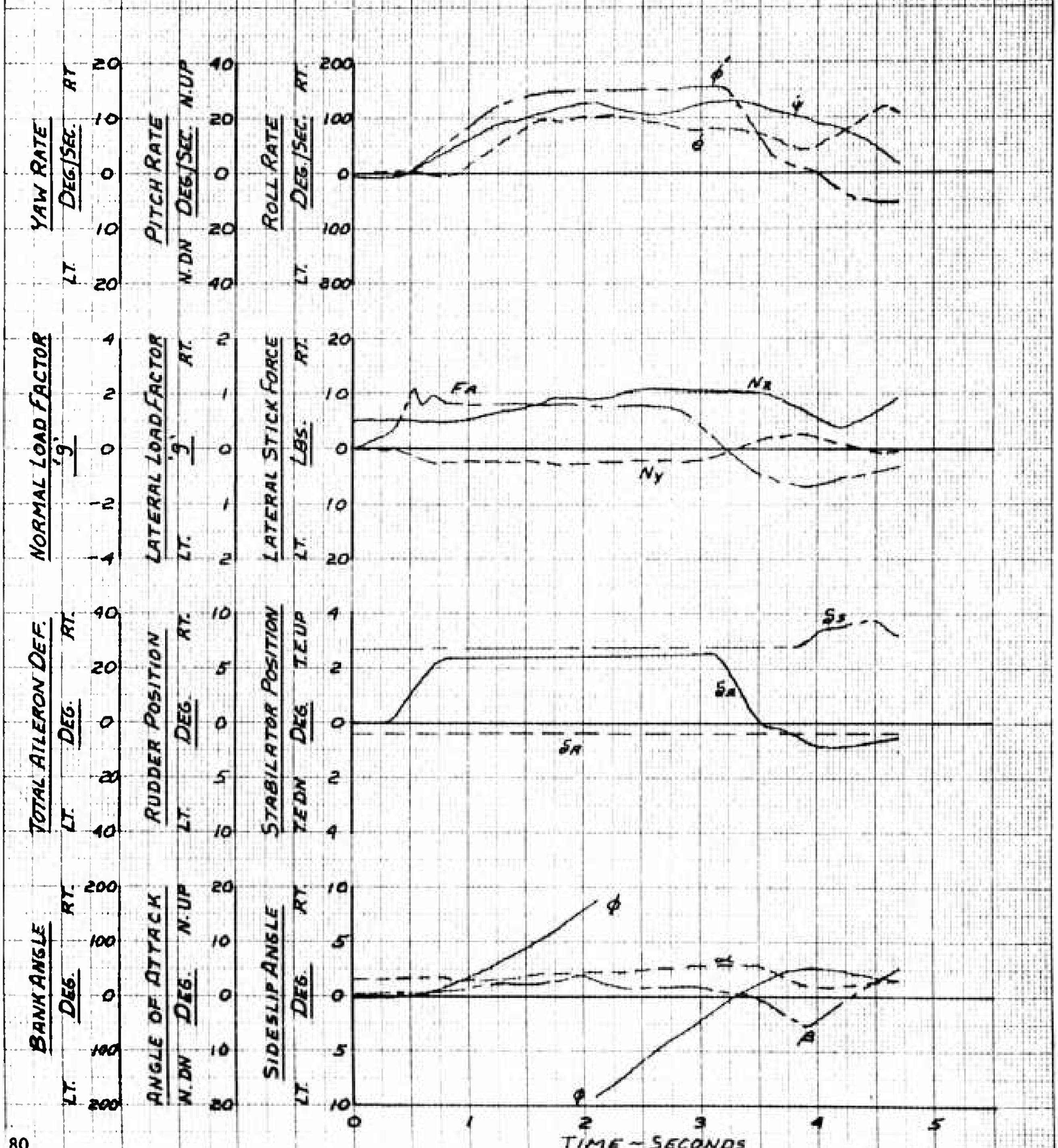


FIG. NO. 53
AILERON ROLLS
T-38A SN58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION
TRIM VC KTS. 243
HP FT 41910
MACH NO. .886
GROSS WT. LBS. 9180
C.G % MAC 15.6
ENTRY'S' 1.0
A.R.I. OUT
AUGS. OFF
YAW RATE DEG/SEC. LT. RT.
PITCH RATE DEG./SEC. N.DN N.UP
ROLL RATE DEG./SEC. LT. RT.
NORMAL LOAD FACTOR g's
LATERAL LOAD FACTOR g's LT. RT.
LATERAL STICK FORCE LBS. LT. RT.
TOTAL AILERON DEF. DEG. LT. RT.
RUDDER POSITION DEG. LT. RT.
STABILATOR POSITION DEG. T.E.DN T.E.UP
BANK ANGLE DEG. LT. RT.
ANGLE OF ATTACK DEG. N.DN N.UP
SIDESLIP ANGLE DEG. LT. RT.
TIME ~ SECONDS

FIG. NO. 54
AILERON ROLLS
T-38A SN58-1195 YJ85-5 ENGINES
CRUISE CONFIGURATION

TRIM VC KTS.	HP FT	MACH NO.	GROSS WT LBS.	C.G. % MAC.	ENTRY 'g'	A.R.I.	AUGS
305	44470	1.064	10350	15.1	1.0	OUT	OFF



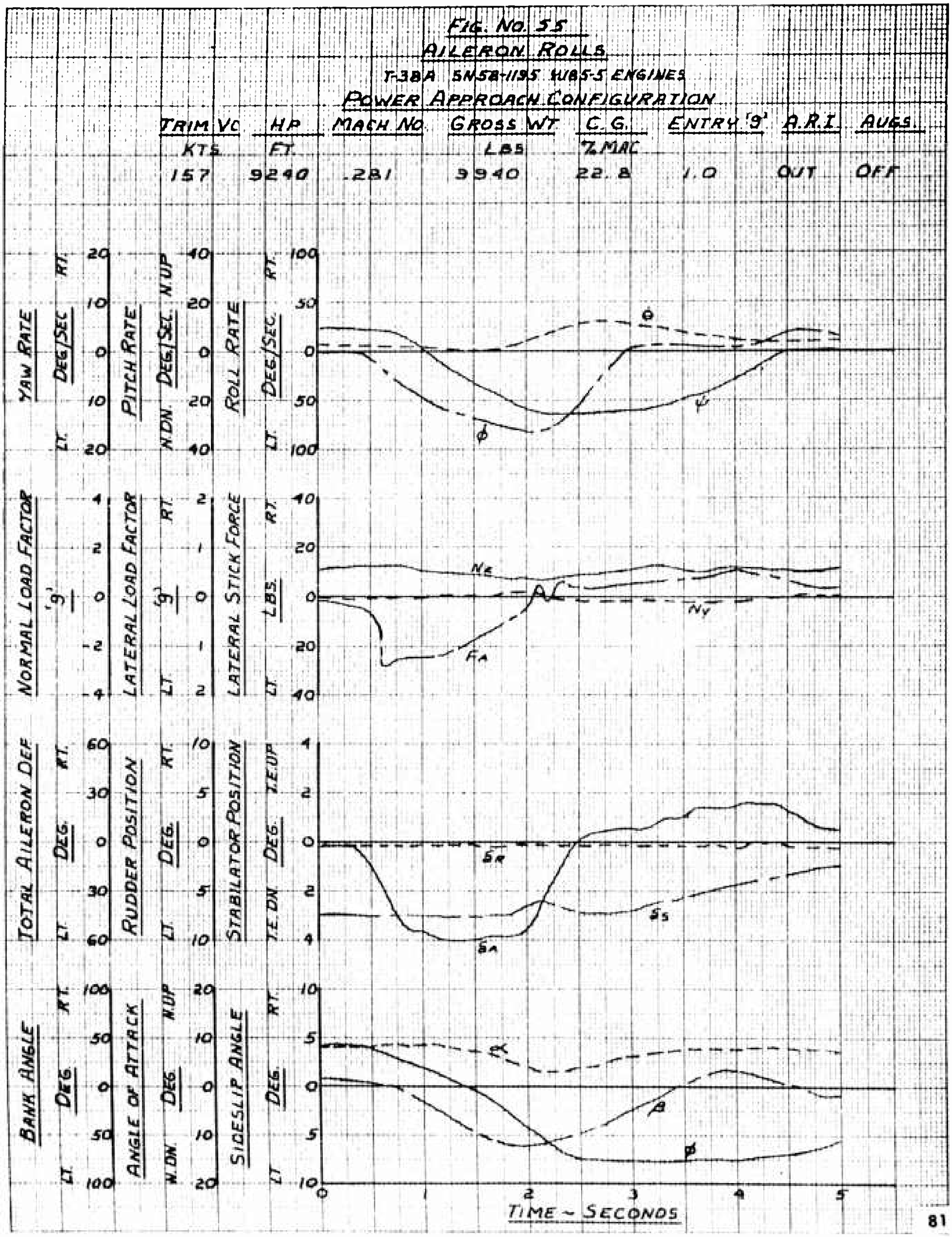


FIG. NO. 55
AILERON ROLLS
T-38A 5N58-1195 J-85-5 ENGINES
POWER APPROACH CONFIGURATION
TRIM VC KTS 157
HP FT. 9240
MACH NO. .281
GROSS WT LBS 9940
C.G. % MAC 22.8
ENTRY 'g' 1.0
A.R.I. OUT
AUGS. OFF
YAW RATE DEG/SEC
PITCH RATE DEG/SEC
ROLL RATE DEG/SEC
NORMAL LOAD FACTOR 'g'
LATERAL LOAD FACTOR 'g'
LATERAL STICK FORCE LBS.
TOTAL AILERON DEF DEG.
RUDDER POSITION DEG.
STABILATOR POSITION DEG.
BANK ANGLE DEG.
ANGLE OF ATTACK DEG.
SIDESLIP ANGLE DEG.
TIME ~ SECONDS

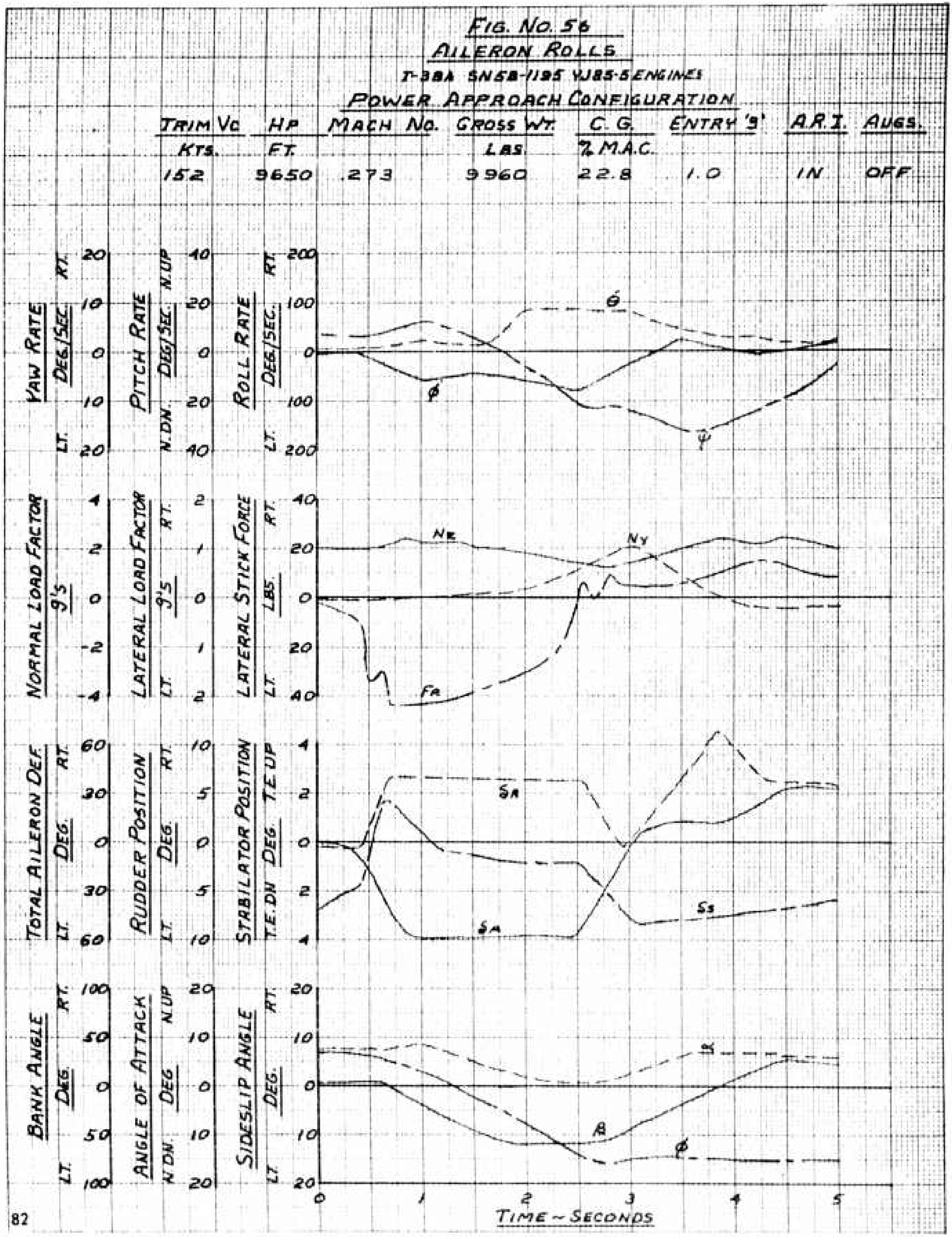


FIG. NO. 56
AILERON ROLLS
T-38A SN58-1195 YJ85-5 ENGINES
POWER APPROACH CONFIGURATION
TRIM VC KTS. 152
HP FT. 9650
MACH NO. .273
GROSS WT. LBS. 9960
C. G. % M.A.C. 22.8
ENTRY 'g' 1.0
A.R.I. IN
AUGS. OFF
YAW RATE DEG/SEC
PITCH RATE DEG/SEC
ROLL RATE DEG/SEC
NORMAL LOAD FACTOR g's
LATERAL LOAD FACTOR g's
LATERAL STICK FORCE LBS.
TOTAL AILERON DEF. DEG.
RUDDER POSITION DEG.
STABILATOR POSITION DEG. T.E. DN T.E. UP
BANK ANGLE DEG.
ANGLE OF ATTACK DEG N.DN N.UP
SIDESLIP ANGLE DEG.
LT. RT.
TIME ~ SECONDS

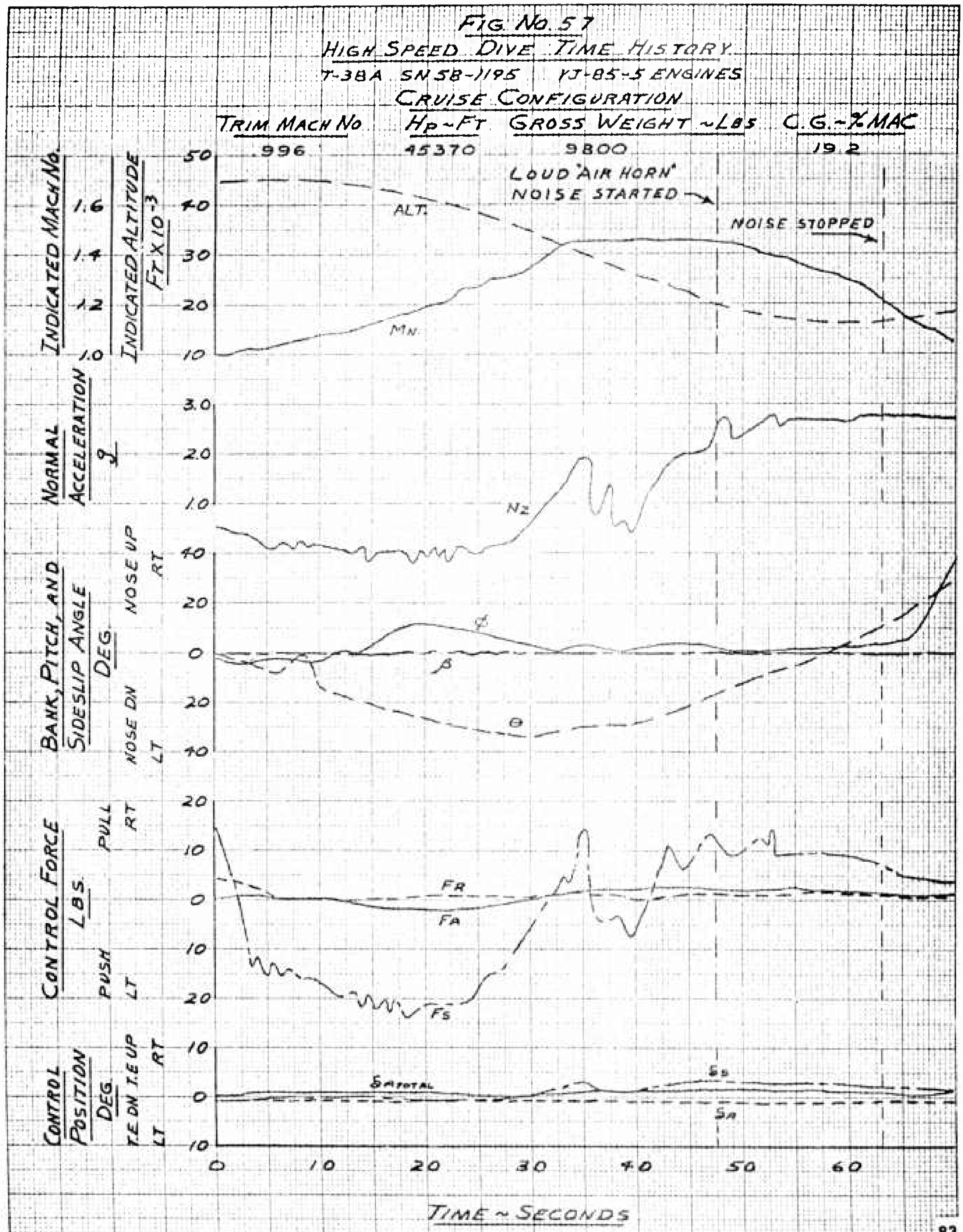


FIG. No. 57
HIGH SPEED DIVE TIME HISTORY
T-38A SN 58-1195 YJ-85-5 ENGINES
CRUISE CONFIGURATION
TRIM MACH NO .996
Hp ~ FT 45370
GROSS WEIGHT ~ LBS 9800
C.G. ~ % MAC 19.2
INDICATED MACH NO.
INDICATED ALTITUDE FT X 10^-3
LOUD "AIR HORN" NOISE STARTED
NOISE STOPPED
ALT.
MN
NORMAL ACCELERATION g
NZ
BANK, PITCH, AND SIDESLIP ANGLE DEG.
NOSE UP RT
NOSE DN LT
φ
β
θ
CONTROL FORCE LBS.
PULL RT
PUSH LT
FR
FA
FS
CONTROL POSITION DEG
T.E UP RT
T.E DN LT
δA TOTAL
δS
δR
TIME ~ SECONDS

FIG. NO. 58

STALL SPEED SUMMARY

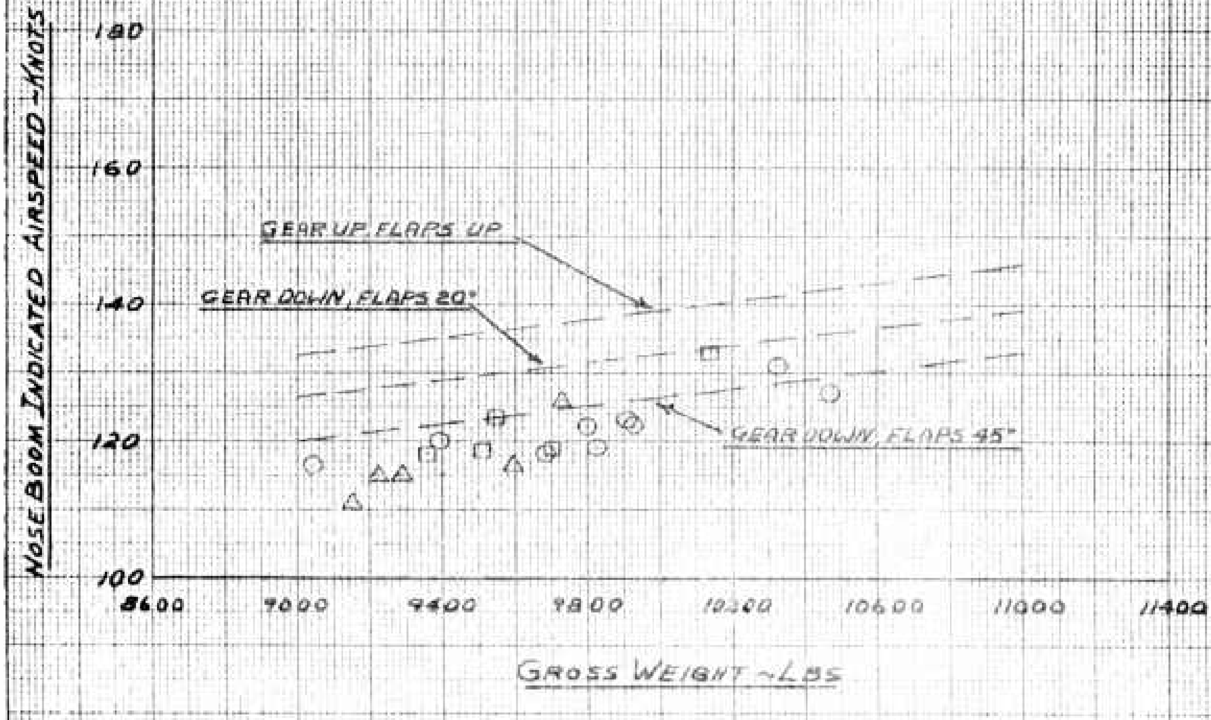
T-38A SN 58-1195 YJ-85-5 ENGINES

APPROXIMATELY 10000 FEET

SYMBOL	CONFIGURATION
○	CRUISE
□	TAKE-OFF
△	LANDING

NOTE: (1) DASHED LINES ARE DATA BASED ON SPEED AT WHICH A RATE OF SINK STARTS WITH POWER FOR LEVEL FLIGHT AT TRIM SPEED OF APPROXIMATELY 1.4 V_s OR V_{max}, WHICHEVER IS LESS

(2) DATA POINTS AND SOLID LINE ARE MAXIMUM AIRSPEED HIGH RATE OF SINK DATA.



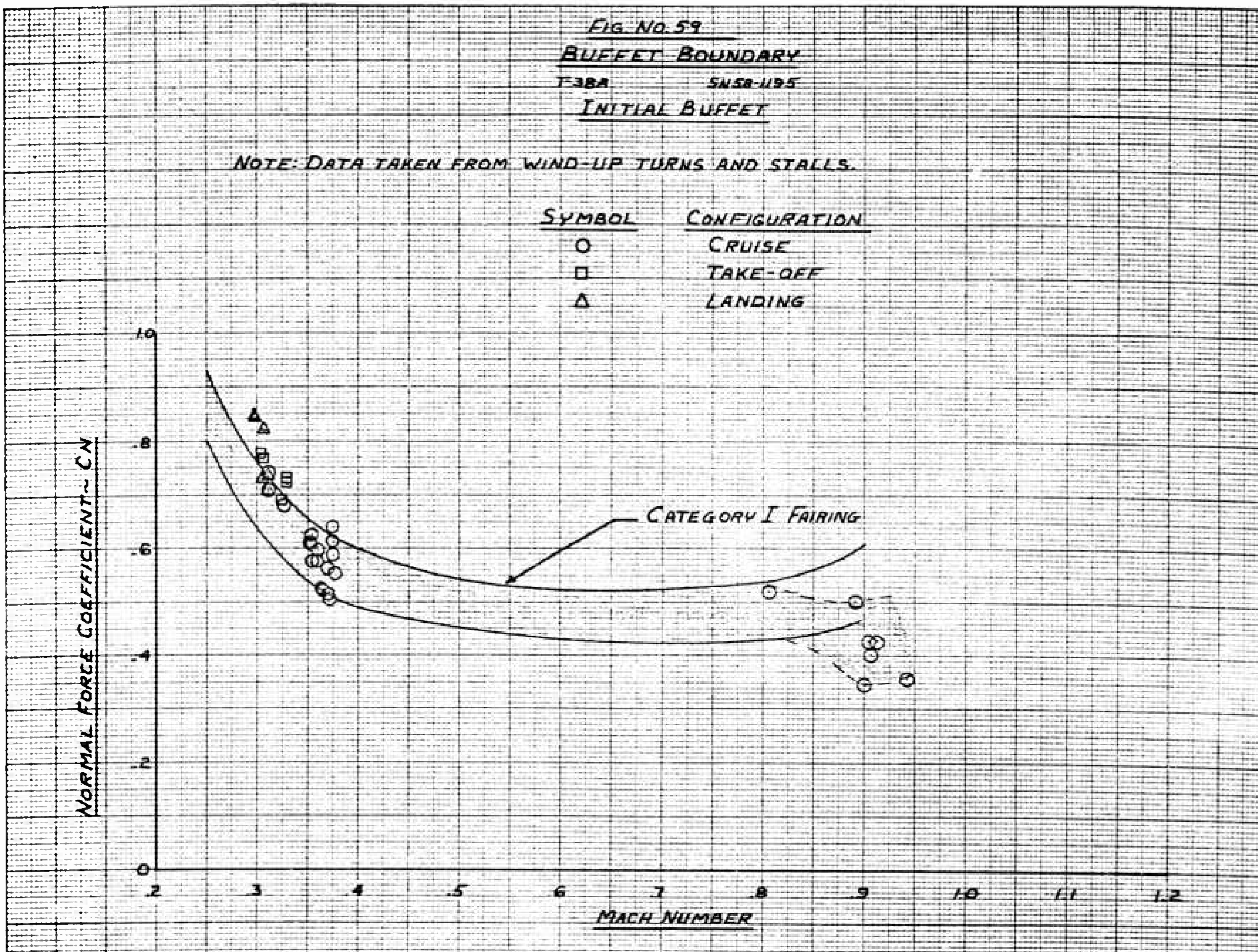


FIG. NO. 59
BUFFET BOUNDARY
T-38A
SN58-1195
INITIAL BUFFET
NOTE: DATA TAKEN FROM WIND-UP TURNS AND STALLS.
SYMBOL
CONFIGURATION
CRUISE
TAKE-OFF
LANDING
CATEGORY I FAIRING
NORMAL FORCE COEFFICIENT ~ CN
MACH NUMBER
1.0
.8
.6
.4
.2
0
.2
.3
.4
.5
.6
.7
.8
.9
1.0
1.1
1.2

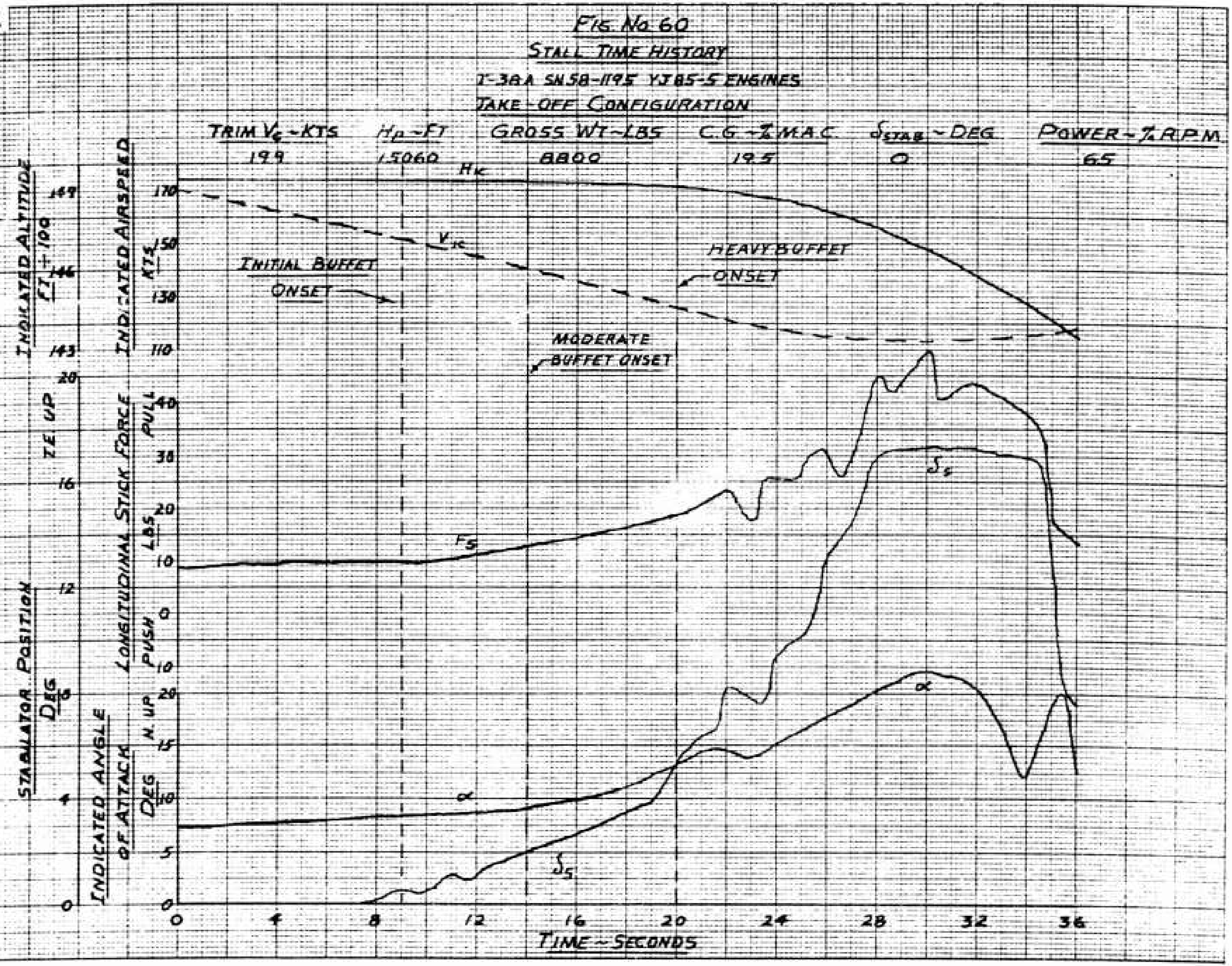


FIG. NO 60
STALL TIME HISTORY
T-38A SN58-1195 YJ85-5 ENGINES
TAKE-OFF CONFIGURATION
TRIM V_c ~ KTS 199
H_p ~ FT 15060
GROSS WT ~ LBS 8800
C.G. ~ % M.A.C. 19.5
δ_{STAB} ~ DEG 0
POWER ~ % R.P.M. 65
INITIAL BUFFET ONSET
MODERATE BUFFET ONSET
HEAVY BUFFET ONSET
H_{ic}
V_{ic}
F_s
δ_s
α
INDICATED ALTITUDE FT ÷ 100
INDICATED AIRSPEED KTS
STABILATOR POSITION DEG
T.E. UP
LONGITUDINAL STICK FORCE LBS
PULL
PUSH
INDICATED ANGLE OF ATTACK DEG
N. UP
TIME ~ SECONDS

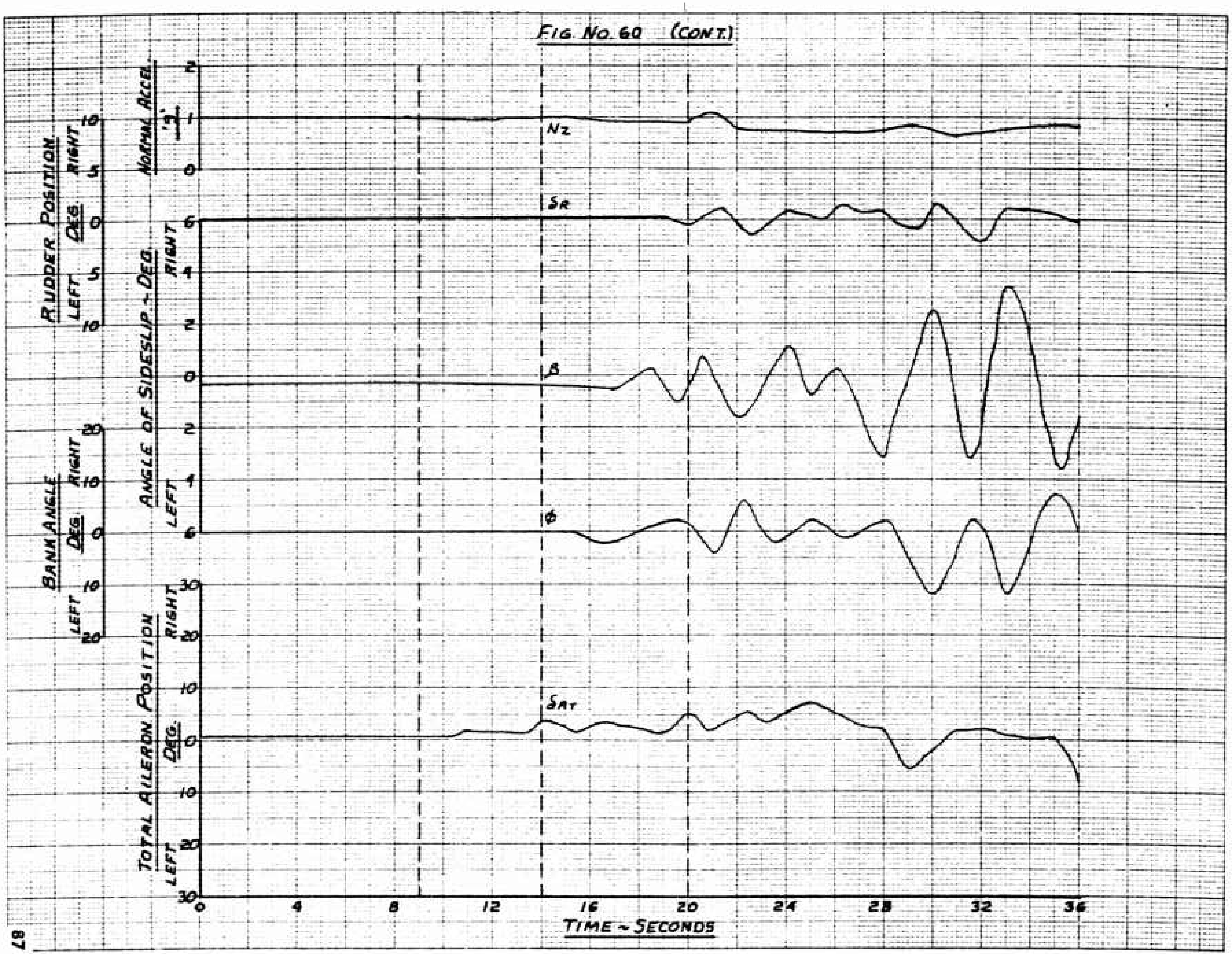


Fig. No. 60 (Cont.)

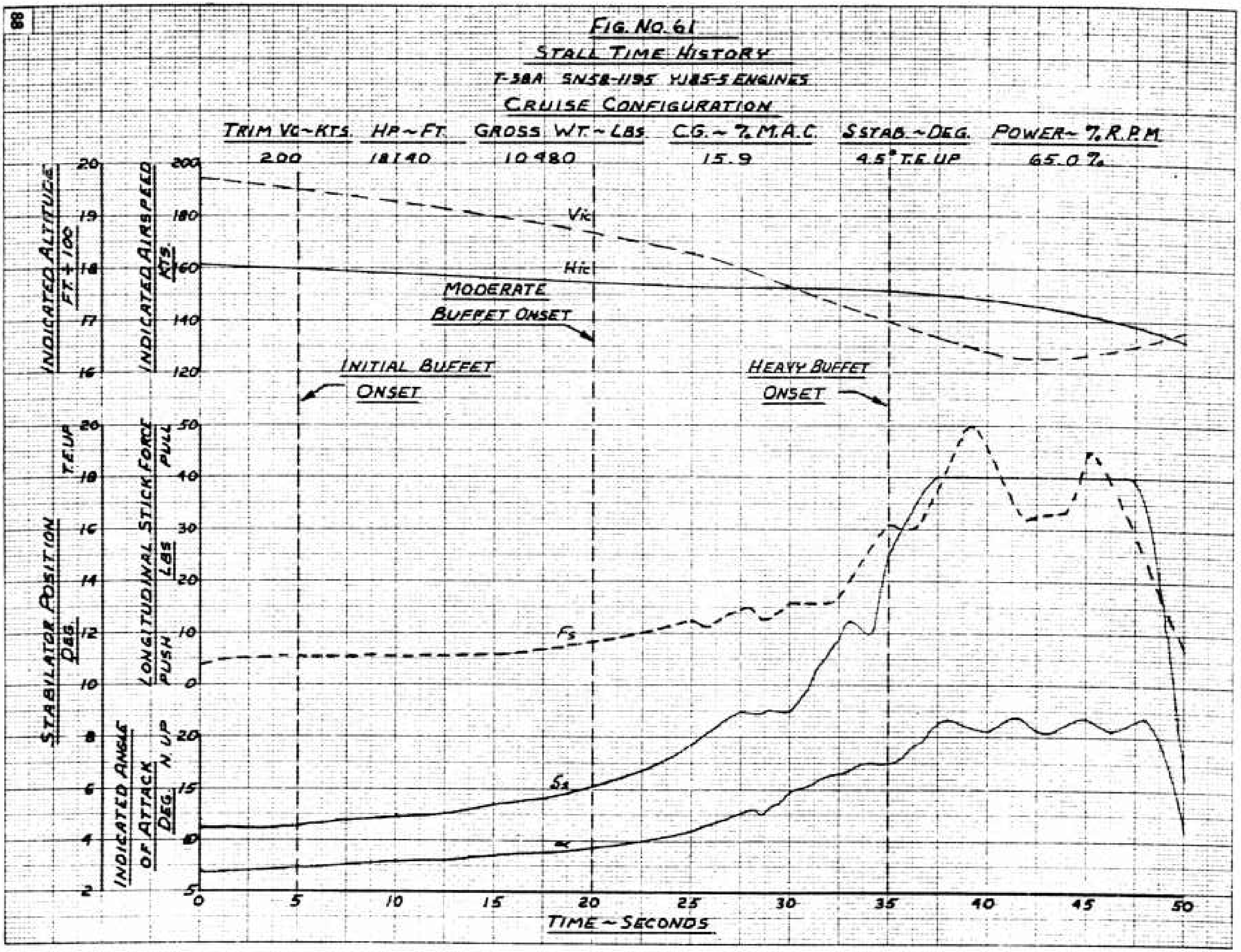


Fig. No. 61
Stall Time History
T-38A SN58-1195 YJ85-5 Engines
Cruise Configuration
Trim Vc~Kts. 200
Hp~Ft. 18140
Gross Wt.~Lbs 10980
C.G.~% M.A.C. 15.9
δ stab.~Deg. 4.5° T.E. Up
Power~% R.P.M. 65.0%
Indicated Altitude Ft. ÷ 100
Indicated Airspeed Kts.
Vic
Hic
Moderate Buffet Onset
Initial Buffet Onset
Heavy Buffet Onset
Stabilator Position Deg. T.E. Up
Longitudinal Stick Force Lbs Push Pull
Fs
δs
α
Indicated Angle of Attack Deg. N. Up
Time ~ Seconds

FIG. NO. 61 (CON'T)

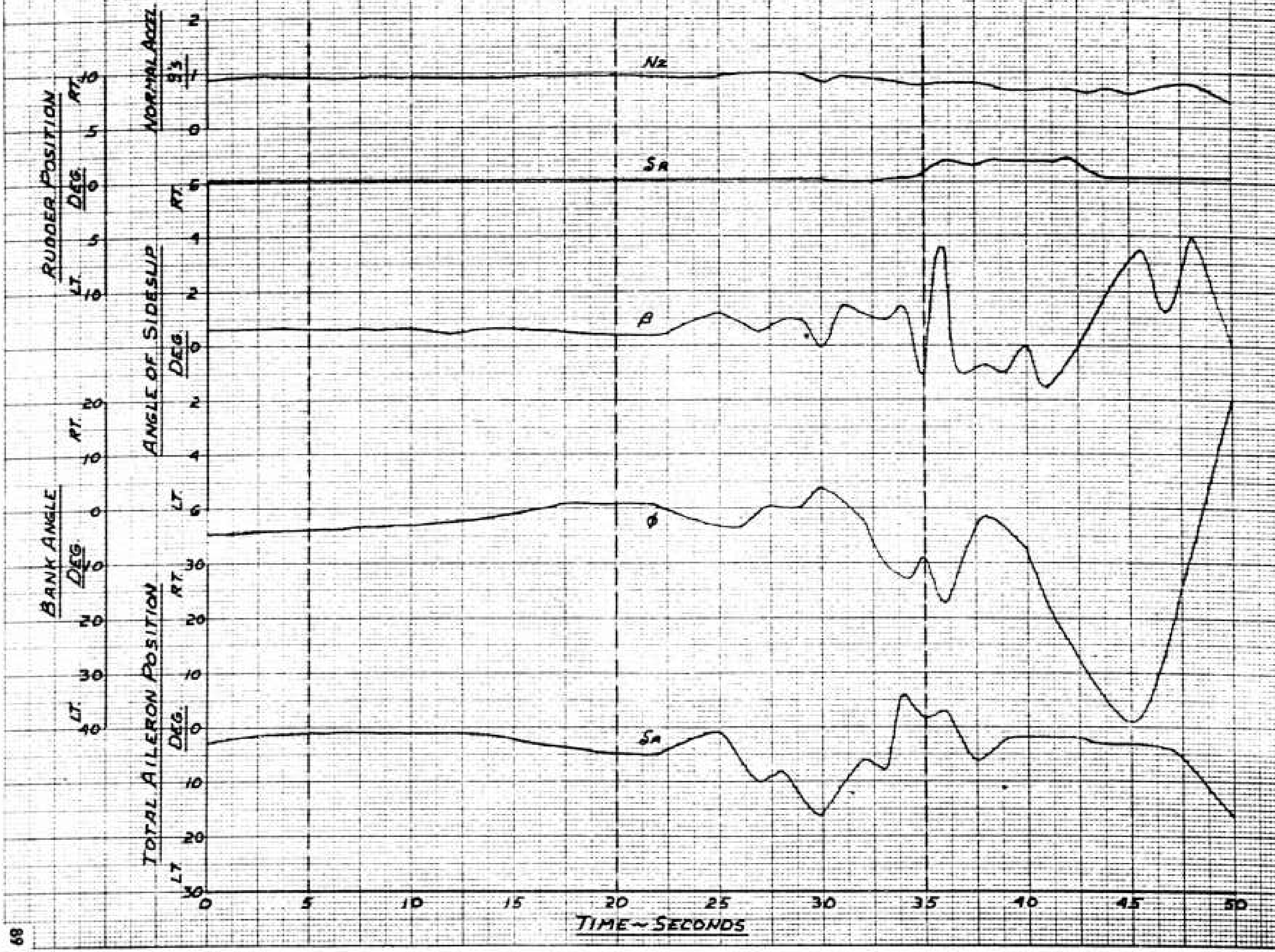


FIG. NO. 62

STALL TIME HISTORY

T-38A SN58-1195 XJ85-5 ENGINES

CRUISE CONFIGURATION

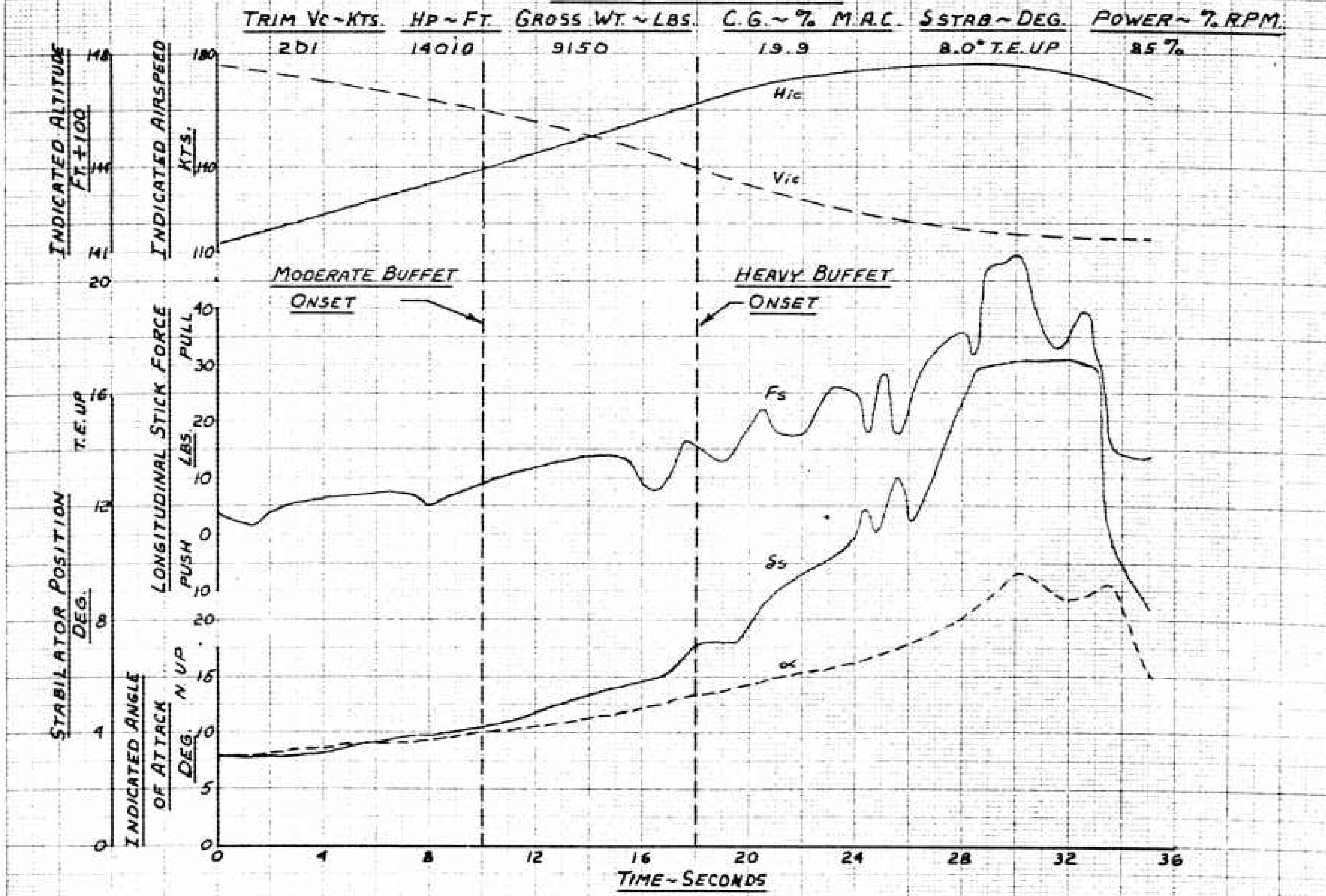
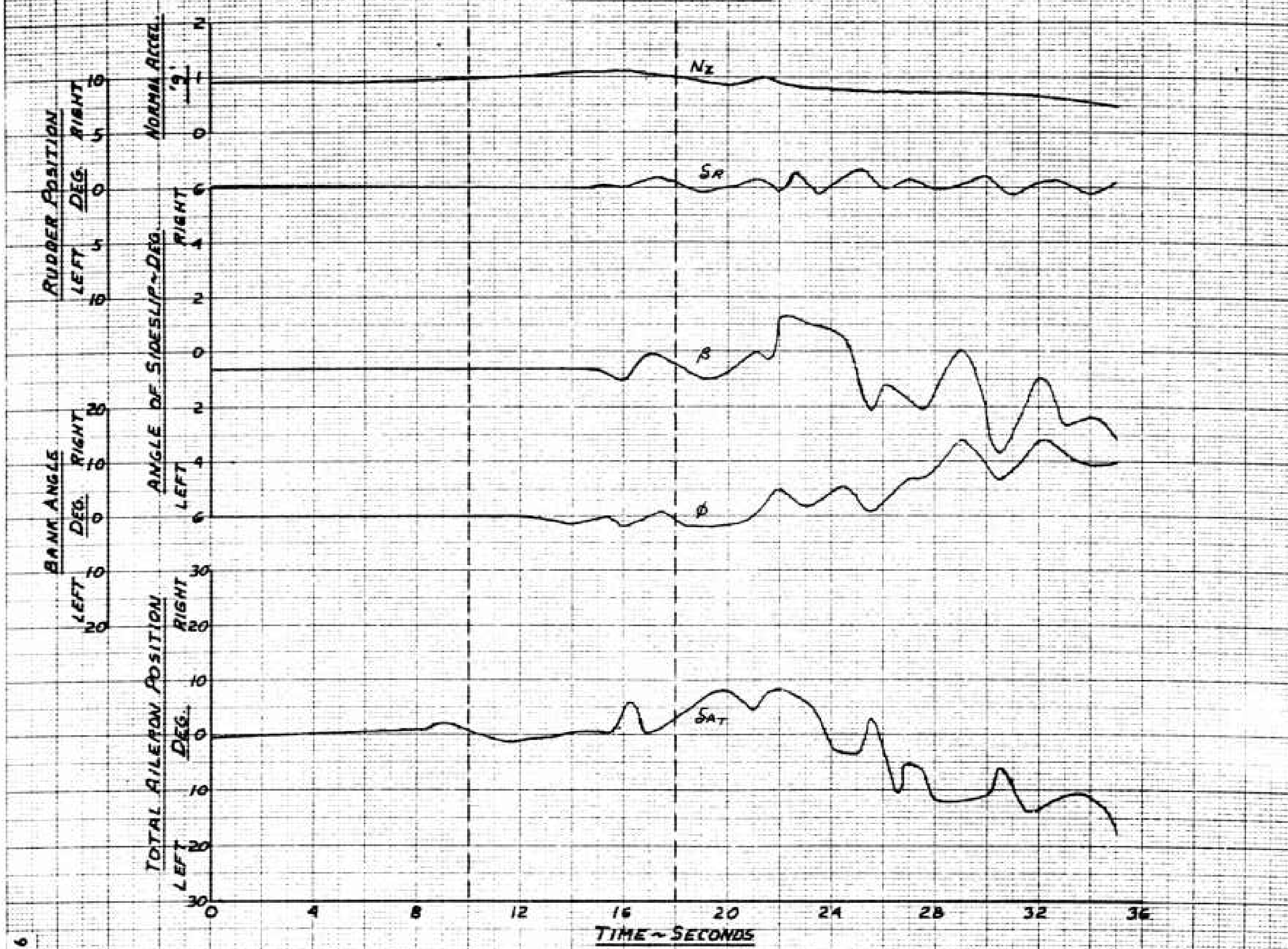


FIG NO. 62 (CONT.)



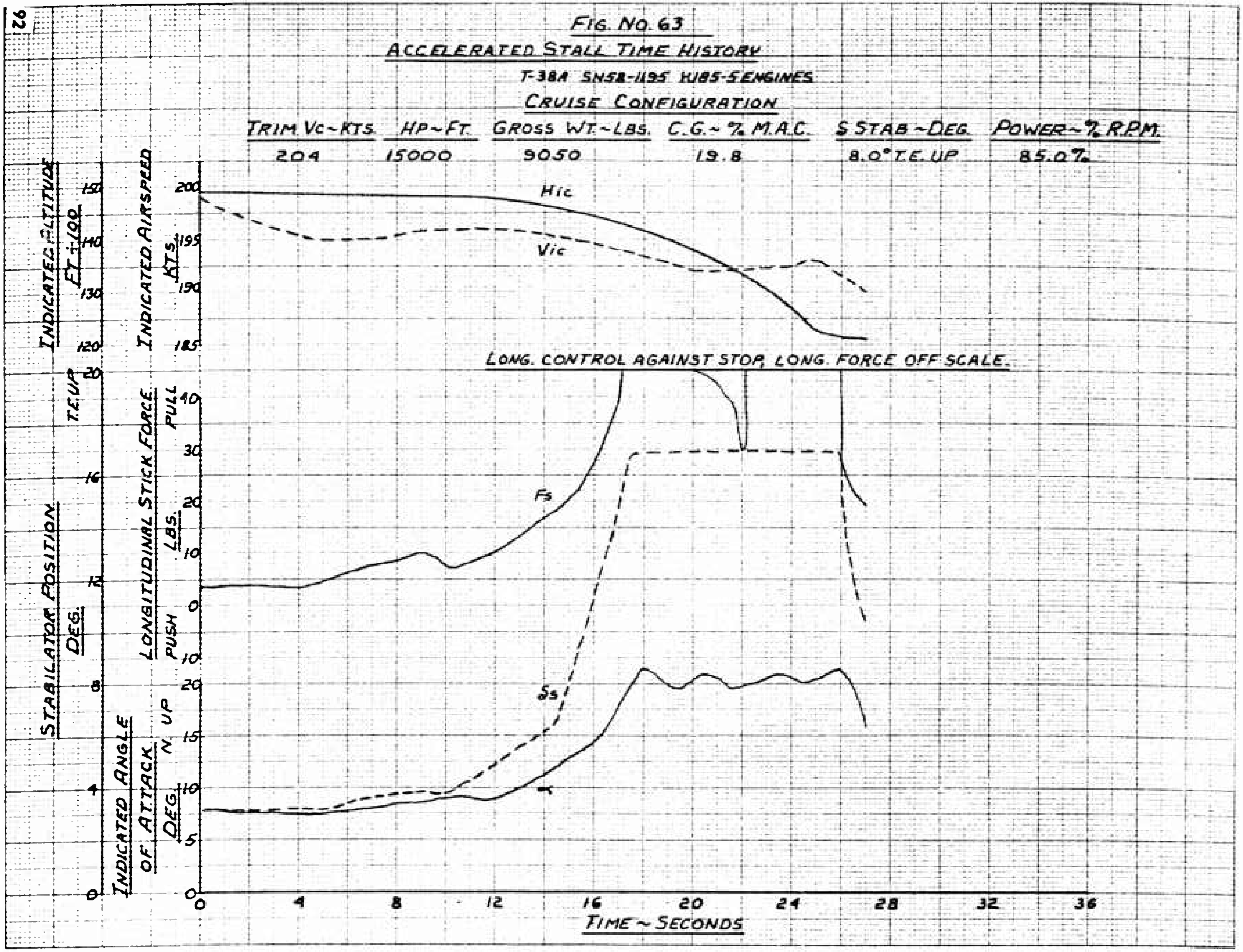
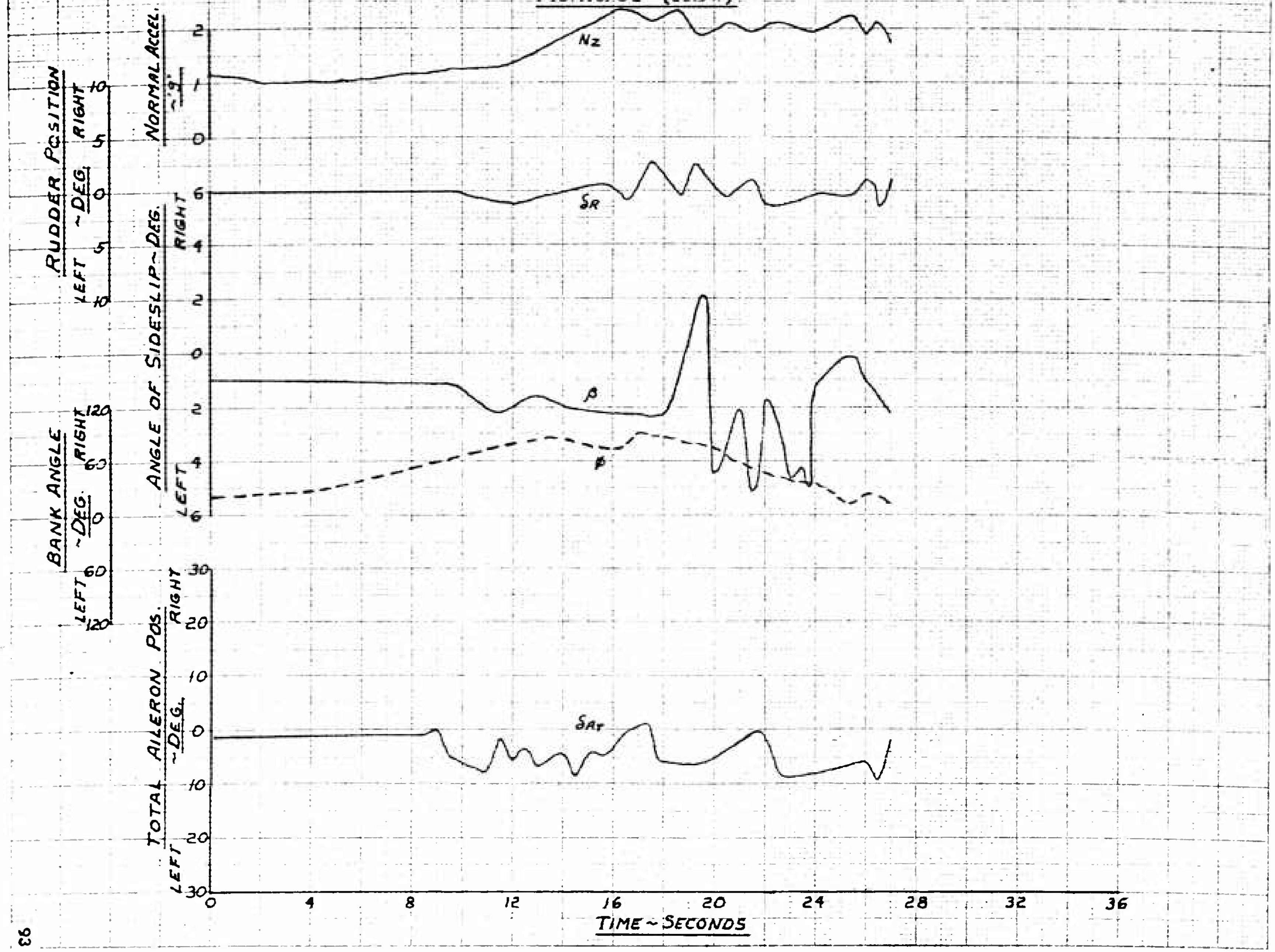


FIG. NO. 63
ACCELERATED STALL TIME HISTORY
T-38A SN58-1195 J85-5 ENGINES
CRUISE CONFIGURATION
TRIM Vc~KTS. 204
HP~FT. 15000
GROSS WT.~LBS. 9050
C.G.~% M.A.C. 19.8
S STAB~DEG. 8.0° T.E. UP
POWER~% R.P.M. 85.0%
Hic
Vic
LONG. CONTROL AGAINST STOP, LONG. FORCE OFF SCALE.
Fs
δs
α
INDICATED ALTITUDE FT÷100
INDICATED AIRSPEED KTS.
STABILATOR POSITION DEG. T.E. UP
LONGITUDINAL STICK FORCE LBS. PULL PUSH
INDICATED ANGLE OF ATTACK DEG. N. UP
TIME ~ SECONDS

FIG. NO. 63 (CONT.)



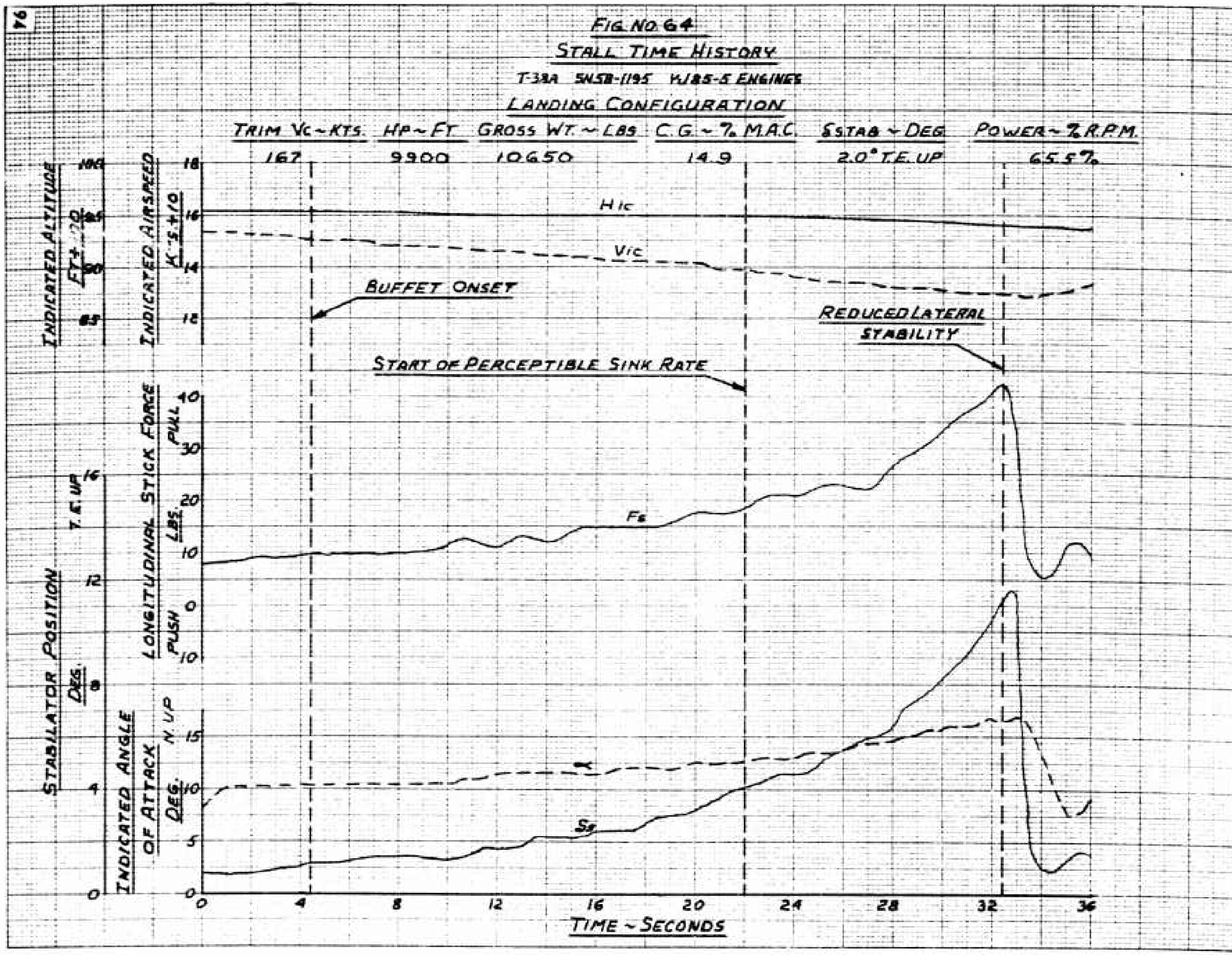


FIG. NO. 64
STALL TIME HISTORY
T-38A SN58-1195 YJ85-5 ENGINES
LANDING CONFIGURATION
TRIM Vc ~ KTS. | HP ~ FT | GROSS WT. ~ LBS | C.G. ~ % M.A.C. | δSTAB ~ DEG. | POWER ~ % R.P.M.
167 | 9900 | 10650 | 14.9 | 2.0° T.E. UP | 65.5%
Hic
Vic
BUFFET ONSET
START OF PERCEPTIBLE SINK RATE
REDUCED LATERAL STABILITY
Fs
δs
INDICATED ALTITUDE FT ÷ 100
INDICATED AIRSPEED K.S. ÷ 10
LONGITUDINAL STICK FORCE LBS.
PULL
PUSH
STABILATOR POSITION DEG. T.E. UP
INDICATED ANGLE OF ATTACK DEG. N. UP
TIME ~ SECONDS

FIG. No. 64 (CONT.)

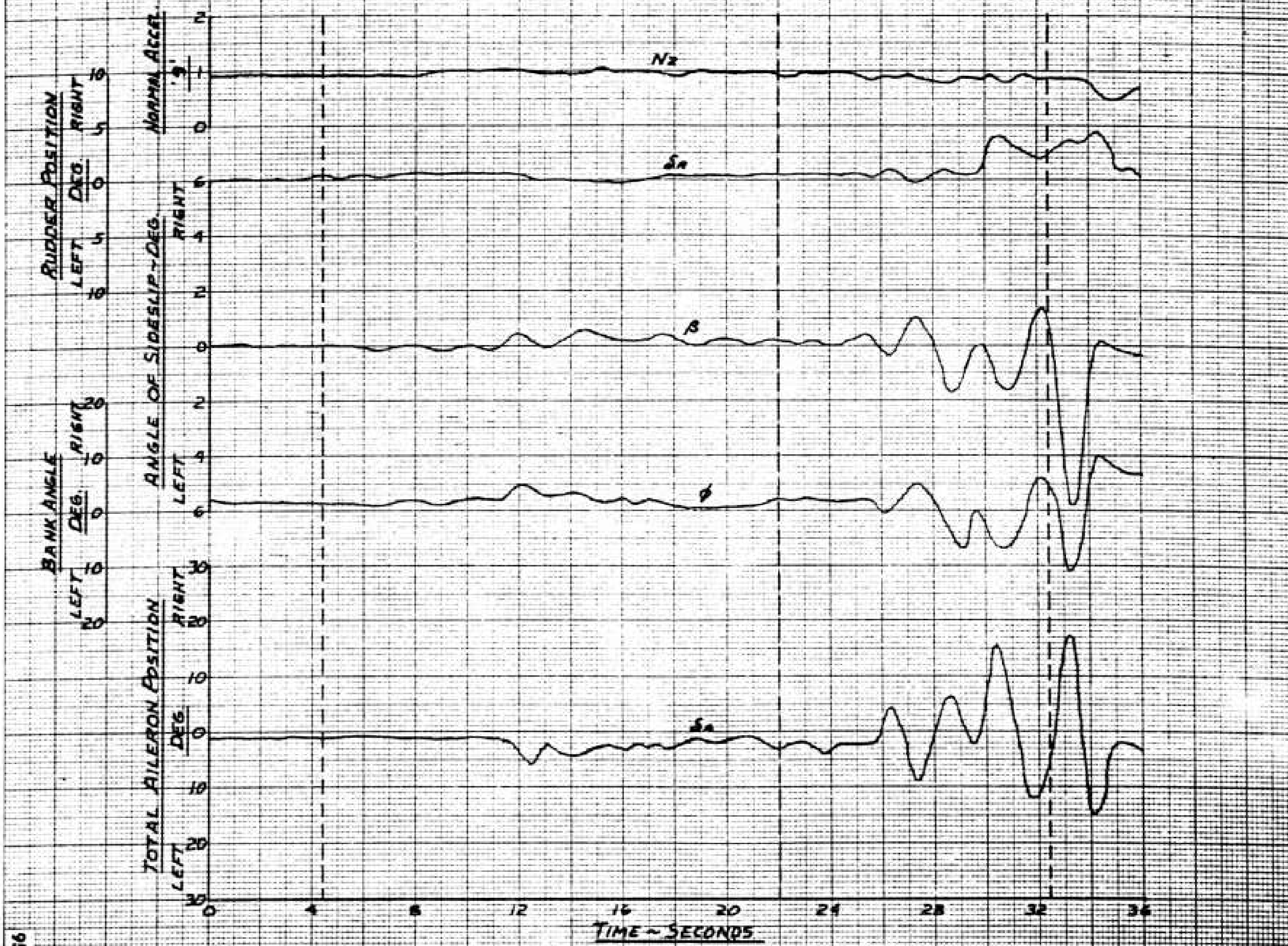
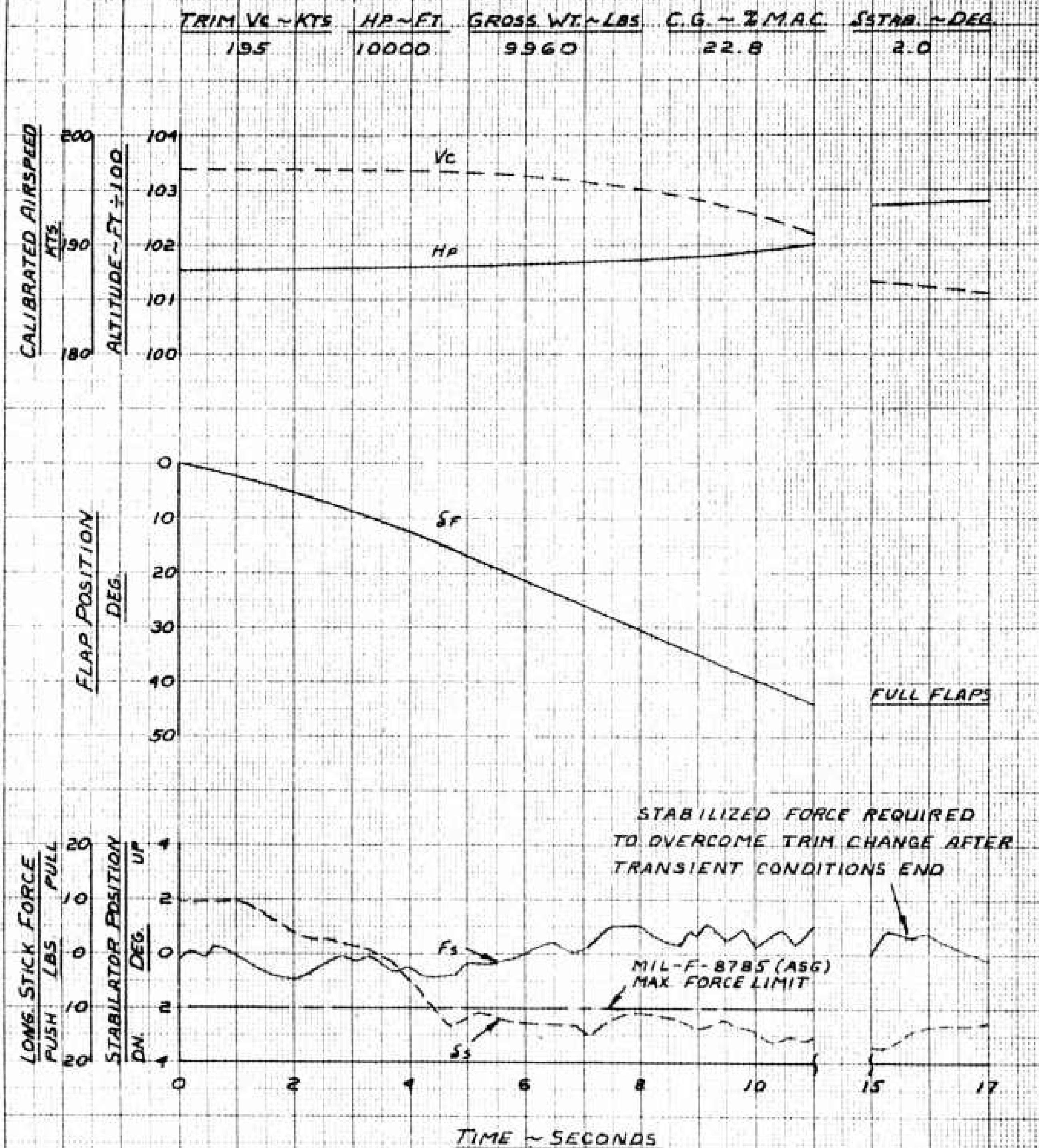


FIG NO 65
TIME HISTORY OF A FLAP EXTENSION
T-38A SN58-1195 YJ85-5 ENGINES



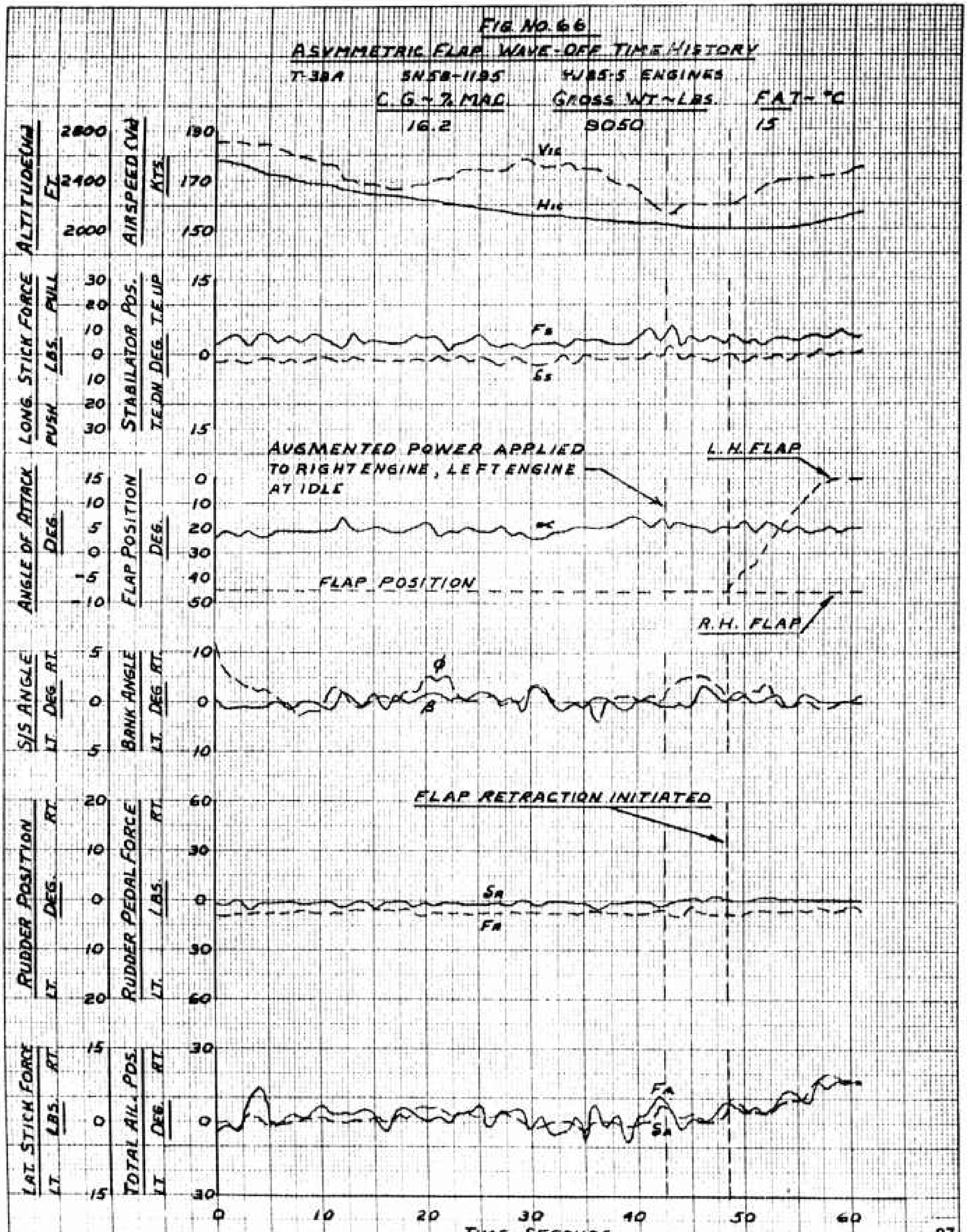


FIG. NO. 66
ASYMMETRIC FLAP WAVE-OFF TIME HISTORY
T-38A
SN 58-1195
J85-5 ENGINES
C.G. ~ % MAC
16.2
GROSS WT ~ LBS.
8050
FAT ~ °C
15
ALTITUDE (Hic) FT.
AIRSPEED (Vic) KTS.
LONG. STICK FORCE PUSH LBS. PULL
STABILATOR POS. T.E. DN DEG. T.E. UP
AUGMENTED POWER APPLIED TO RIGHT ENGINE, LEFT ENGINE AT IDLE
L. H. FLAP
R. H. FLAP
ANGLE OF ATTACK DEG.
FLAP POSITION DEG.
FLAP POSITION
S/S ANGLE LT. DEG. RT.
BANK ANGLE LT. DEG. RT.
FLAP RETRACTION INITIATED
RUDDER POSITION LT. DEG. RT.
RUDDER PEDAL FORCE LT. LBS. RT.
LAT. STICK FORCE LT. LBS. RT.
TOTAL AIL. POS. LT. DEG. RT.
TIME-SECONDS

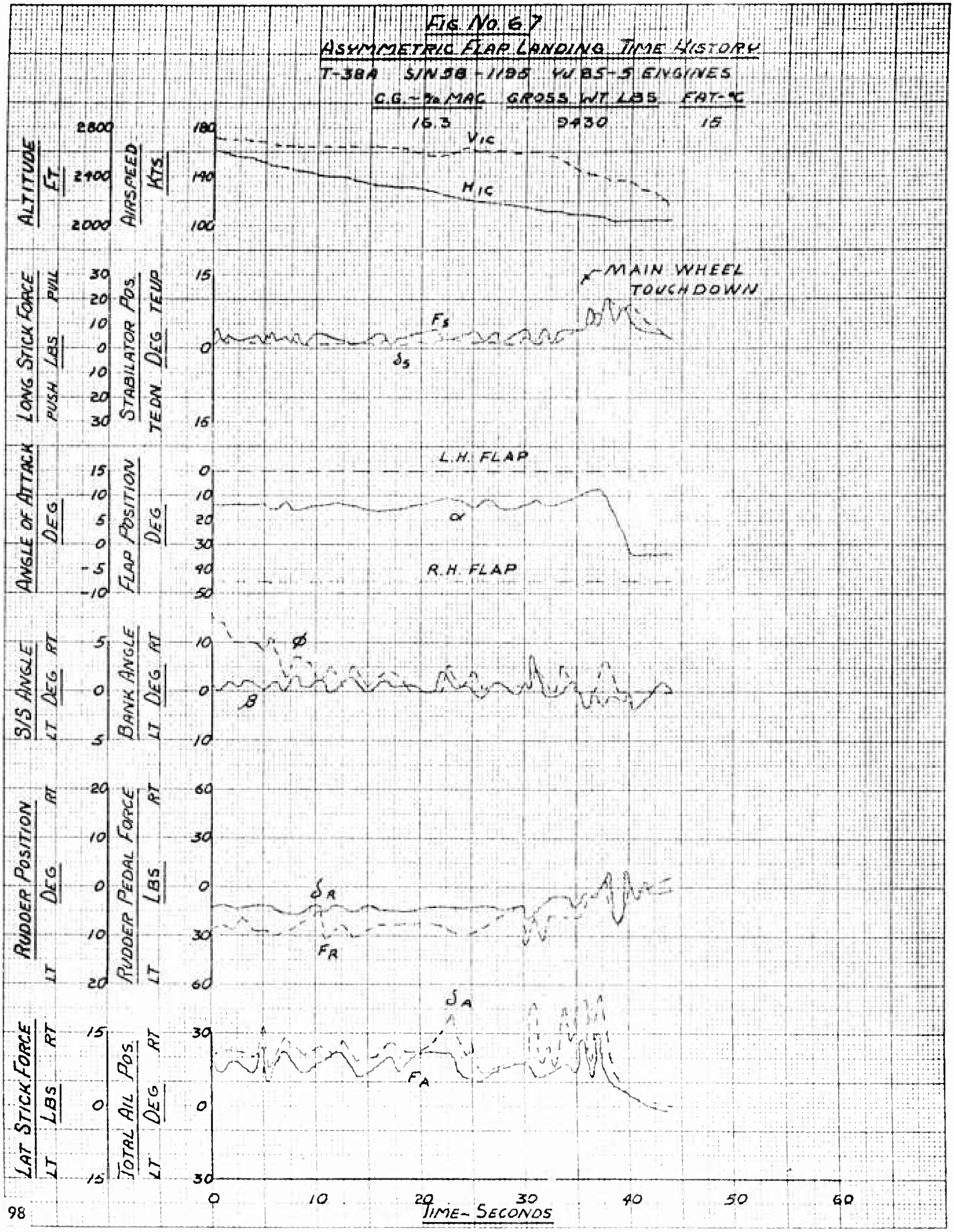


FIG. NO. 67
ASYMMETRIC FLAP LANDING TIME HISTORY
T-38A S/N 58-1195 YJ 85-5 ENGINES
C.G. ~ % MAC GROSS WT. LBS. FAT-°C
16.3 9430 15
ALTITUDE FT
AIRSPEED KTS
V_{IC}
H_{IC}
MAIN WHEEL TOUCHDOWN
LONG STICK FORCE PUSH LBS PULL
STABILATOR POS. TEDN DEG TEUP
F_S
δ_S
ANGLE OF ATTACK DEG
FLAP POSITION DEG
L.H. FLAP
R.H. FLAP
α
S/S ANGLE LT DEG RT
BANK ANGLE LT DEG RT
ϕ
β
RUDDER POSITION LT DEG RT
RUDDER PEDAL FORCE LT LBS RT
δ_R
F_R
LAT STICK FORCE LT LBS RT
TOTAL AIL POS. LT DEG RT
δ_A
F_A
TIME ~ SECONDS

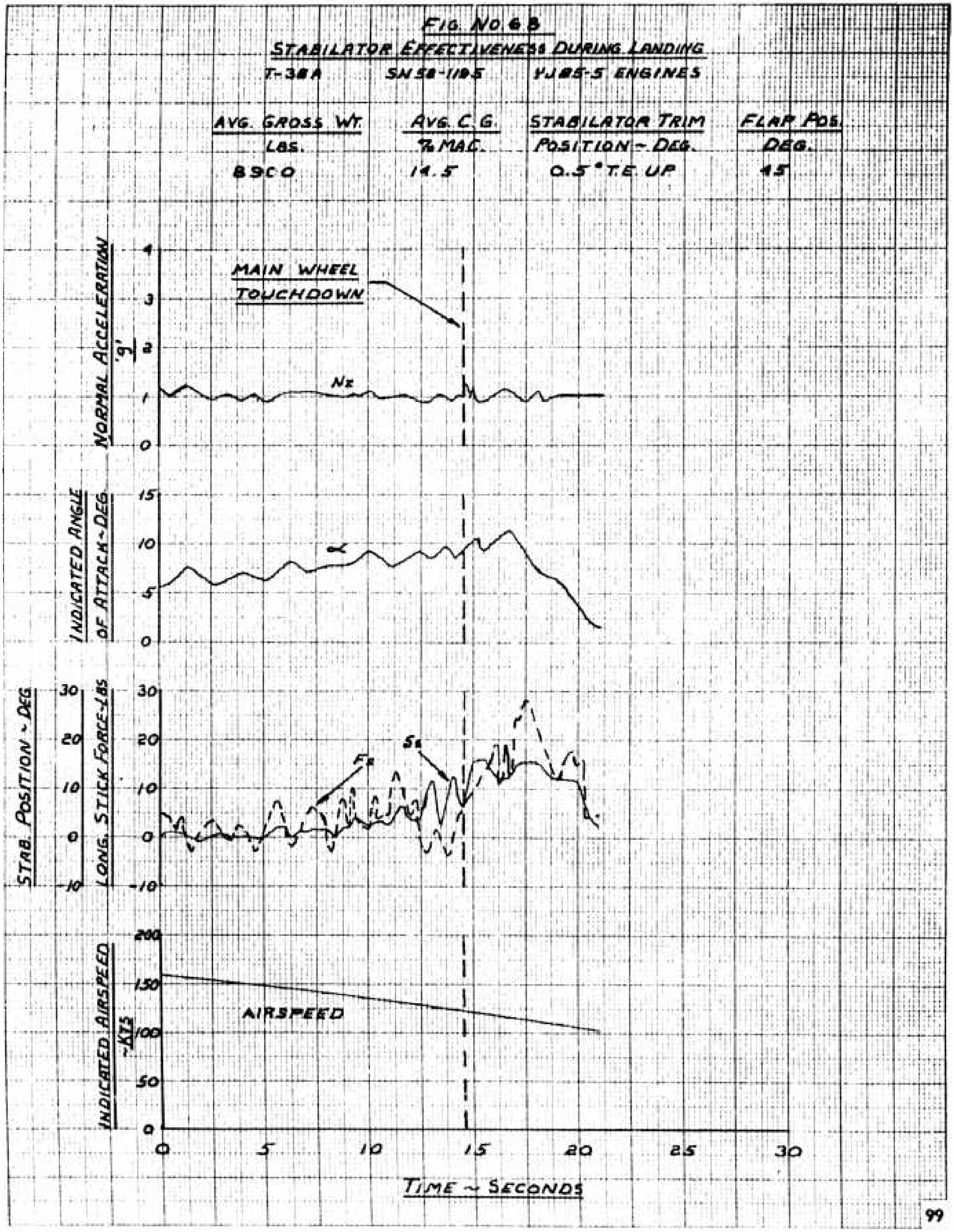


FIG. NO. 6 B
STABILATOR EFFECTIVENESS DURING LANDING
T-38A
SN 58-1195
YJ85-5 ENGINES
AVG. GROSS WT. LBS.
8900
AVG. C.G. % MAC.
14.5
STABILATOR TRIM POSITION ~ DEG.
0.5° T.E. UP
FLAP POS. DEG.
45
MAIN WHEEL TOUCHDOWN
NORMAL ACCELERATION 'g'
Nz
INDICATED ANGLE OF ATTACK ~ DEG.
α
STAB. POSITION ~ DEG
LONG. STICK FORCE-LBS
Fs
Se
INDICATED AIRSPEED ~KTS
AIRSPEED
TIME ~ SECONDS

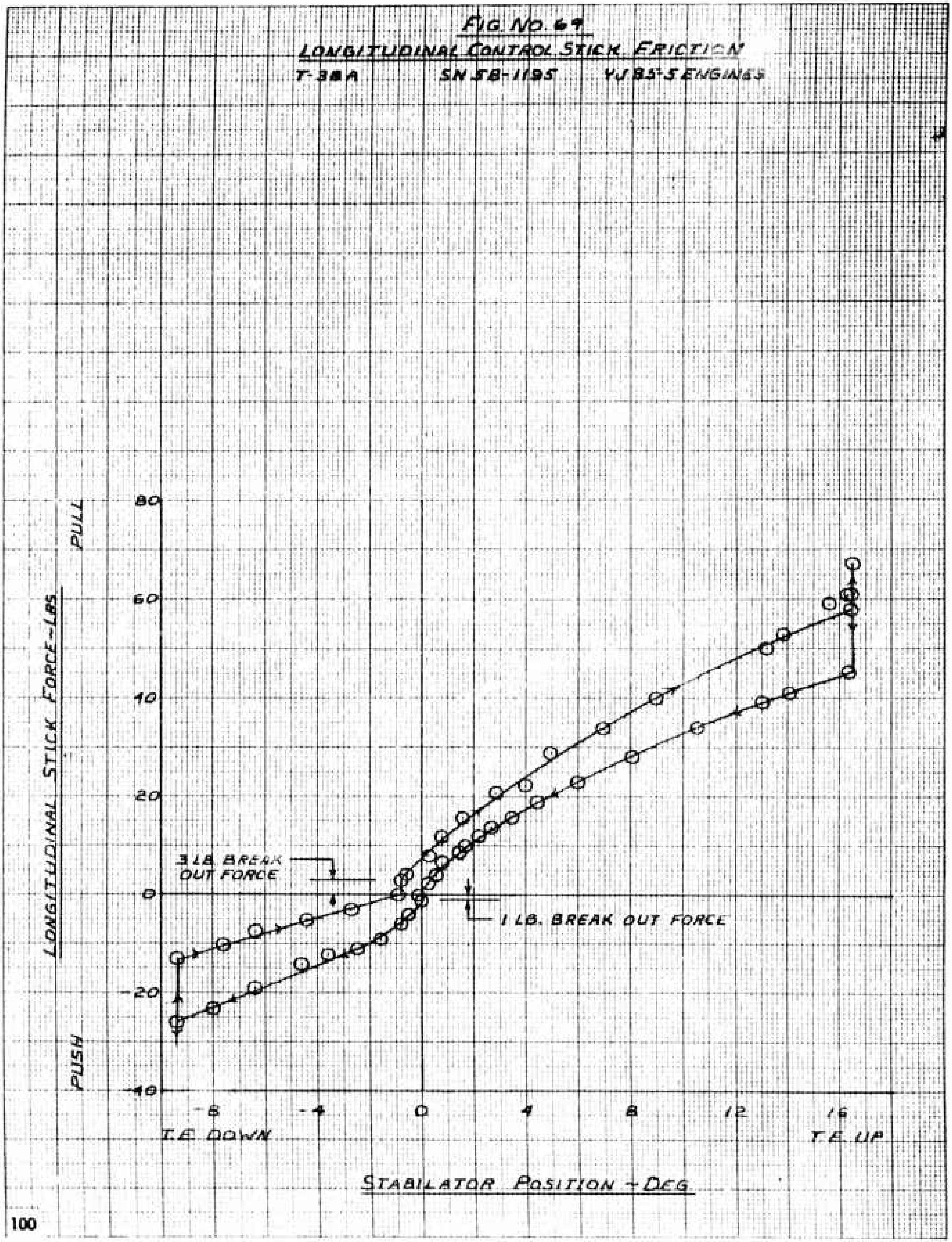
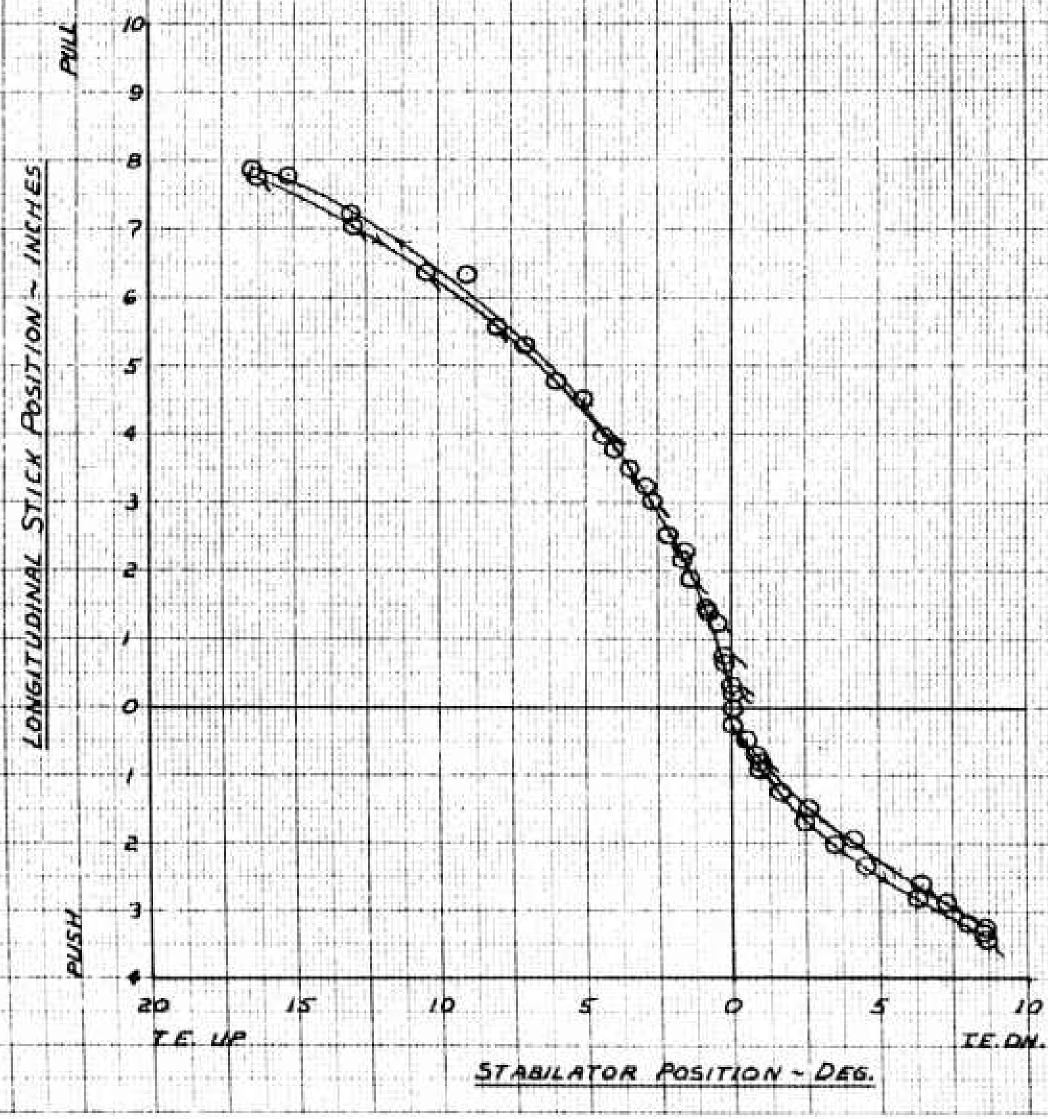


FIG. NO. 69
LONGITUDINAL CONTROL STICK FRICTION
T-38A
SN 58-1195
YJ85-5 ENGINES
LONGITUDINAL STICK FORCE - LBS
PULL
PUSH
80
60
40
20
0
-20
-40
3 LB. BREAK OUT FORCE
1 LB. BREAK OUT FORCE
-8
-4
0
4
8
12
16
T.E. DOWN
T.E. UP
STABILATOR POSITION - DEG

FIG. NO. 70

LONGITUDINAL CONTROL CALIBRATION

T-38A SN 58-1195 YJ85-5 ENGINES



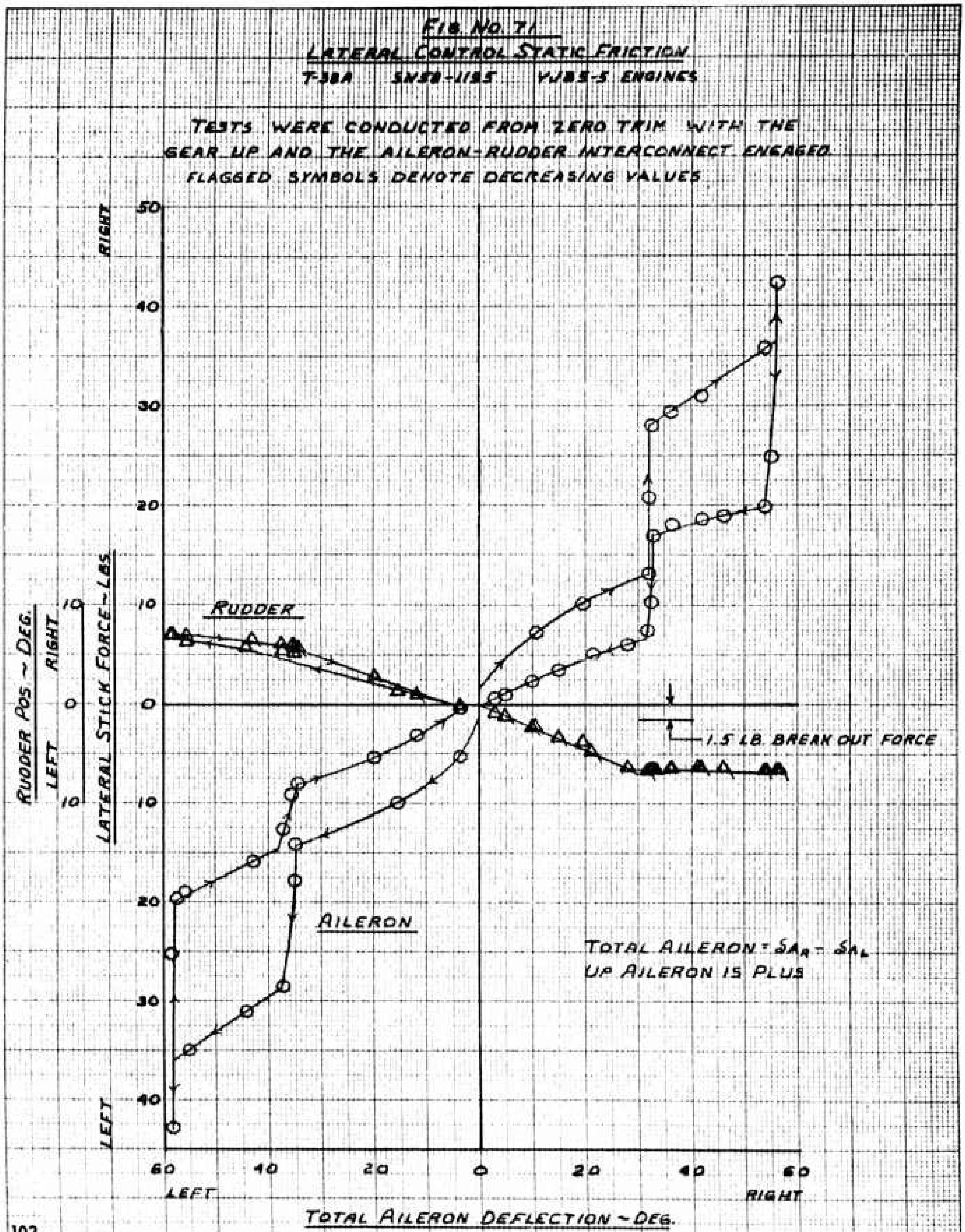


FIG. NO. 71
LATERAL CONTROL STATIC FRICTION
T-38A SN58-1195 YJ85-5 ENGINES
TESTS WERE CONDUCTED FROM ZERO TRIM WITH THE GEAR UP AND THE AILERON-RUDDER INTERCONNECT ENGAGED
FLAGGED SYMBOLS DENOTE DECREASING VALUES
RUDDER
AILERON
1.5 LB. BREAK OUT FORCE
TOTAL AILERON = $\delta A_R - \delta A_L$
UP AILERON IS PLUS
RUDDER POS. ~ DEG.
LEFT
RIGHT
LATERAL STICK FORCE ~ LBS.
TOTAL AILERON DEFLECTION ~ DEG.
LEFT
RIGHT
50
40
30
20
10
0
10
20
30
40
60
40
20
0
20
40
60

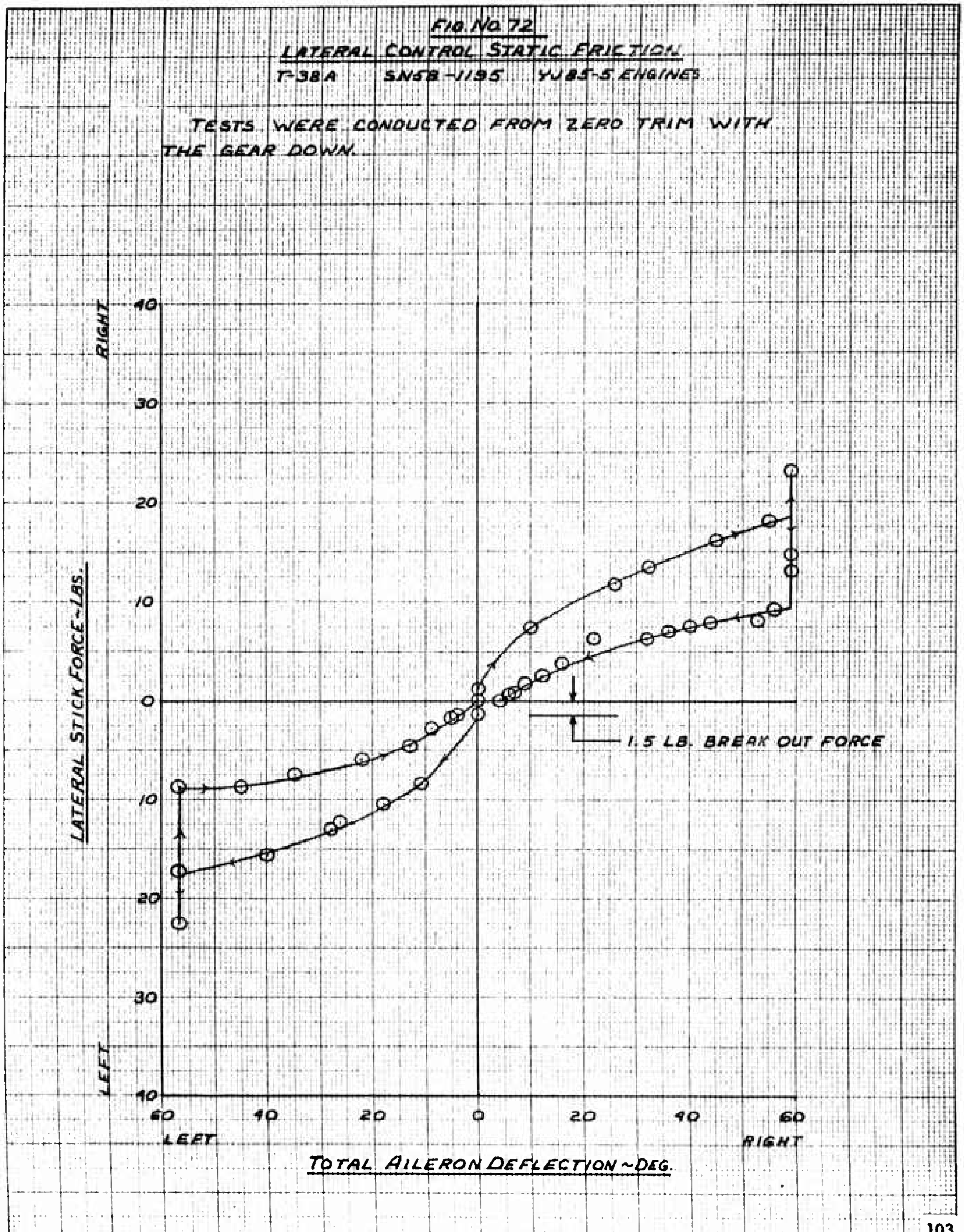


FIG. No. 72
LATERAL CONTROL STATIC FRICTION
T-38A SN58-1195 YJ85-5 ENGINES
TESTS WERE CONDUCTED FROM ZERO TRIM WITH THE GEAR DOWN.
LATERAL STICK FORCE ~ LBS.
RIGHT
LEFT
40
30
20
10
0
10
20
30
40
1.5 LB. BREAK OUT FORCE
60
40
20
0
20
40
60
LEFT
RIGHT
TOTAL AILERON DEFLECTION ~ DEG.

FIG. NO. 73

DIRECTIONAL CONTROL STATIC FRICTION

T-38A SN 58-1195 YJ85-5 ENGINES

NOTE:

1. TESTS WERE CONDUCTED WITH ZERO TRIM AND YAW AUGMENTER OFF AT A HANGER TEMPERATURE OF 70° F.

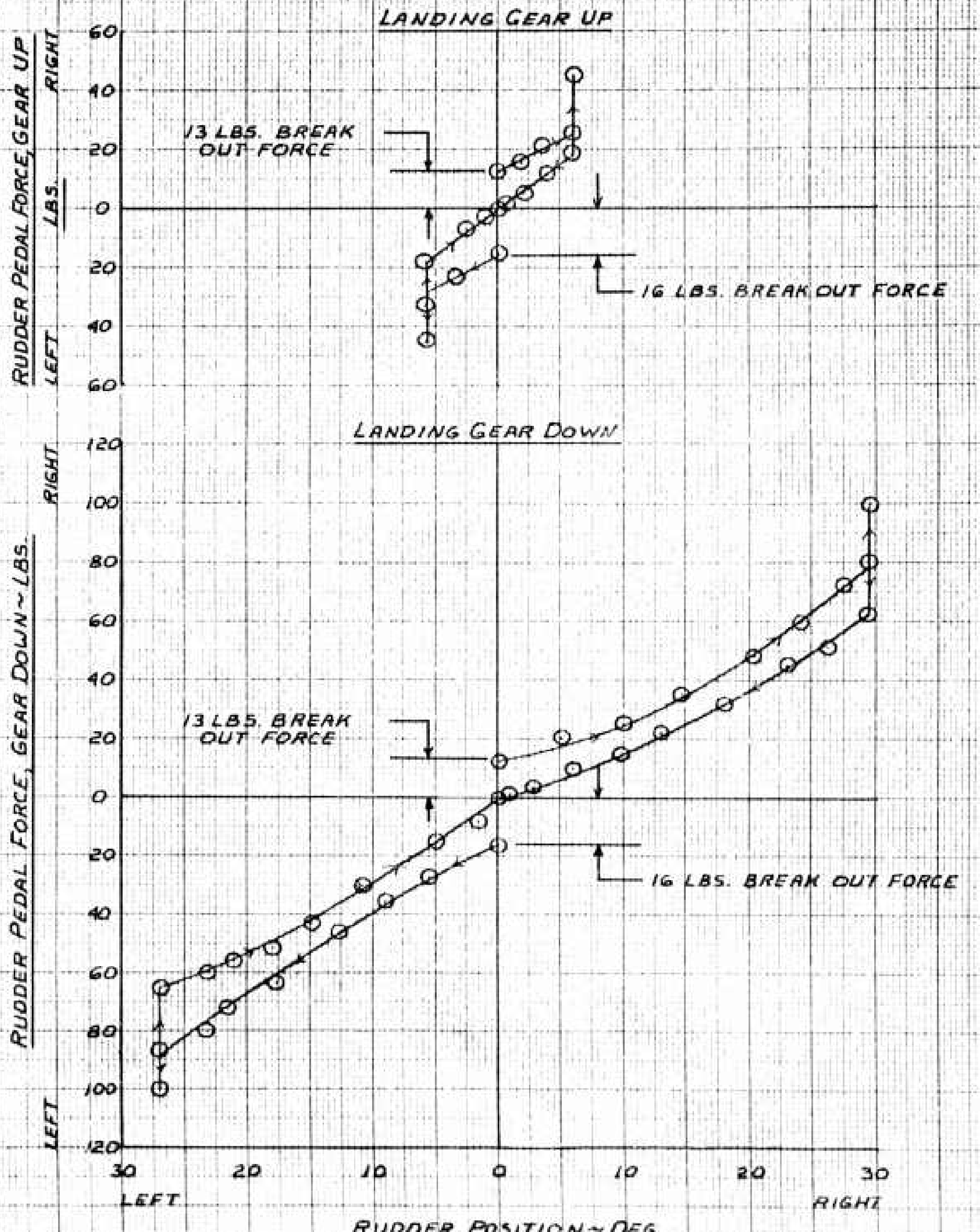


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FIG. NO. 74

TEST BOOM AIRSPEED CALIBRATION

T-38A SN58-1194 & 58-1195 YJ85-5 ENGINES

SYMBOL	AIRPLANE NO.	PACER	PACER NO.	ALTITUDE	GROSS WT.	CONFIG.
○	58-1195	F-104	748	36000 FT.	9300	CRUISE
□	58-1195	F-104	748	36000 FT.	11000	CRUISE
◇	58-1195	F-104	748	36000 FT.	10500	CRUISE
△	58-1194	TOWER FLY-BY		2300 FT.	11300	CRUISE
△	58-1194	F-104	748	36000 FT.	10600	CRUISE

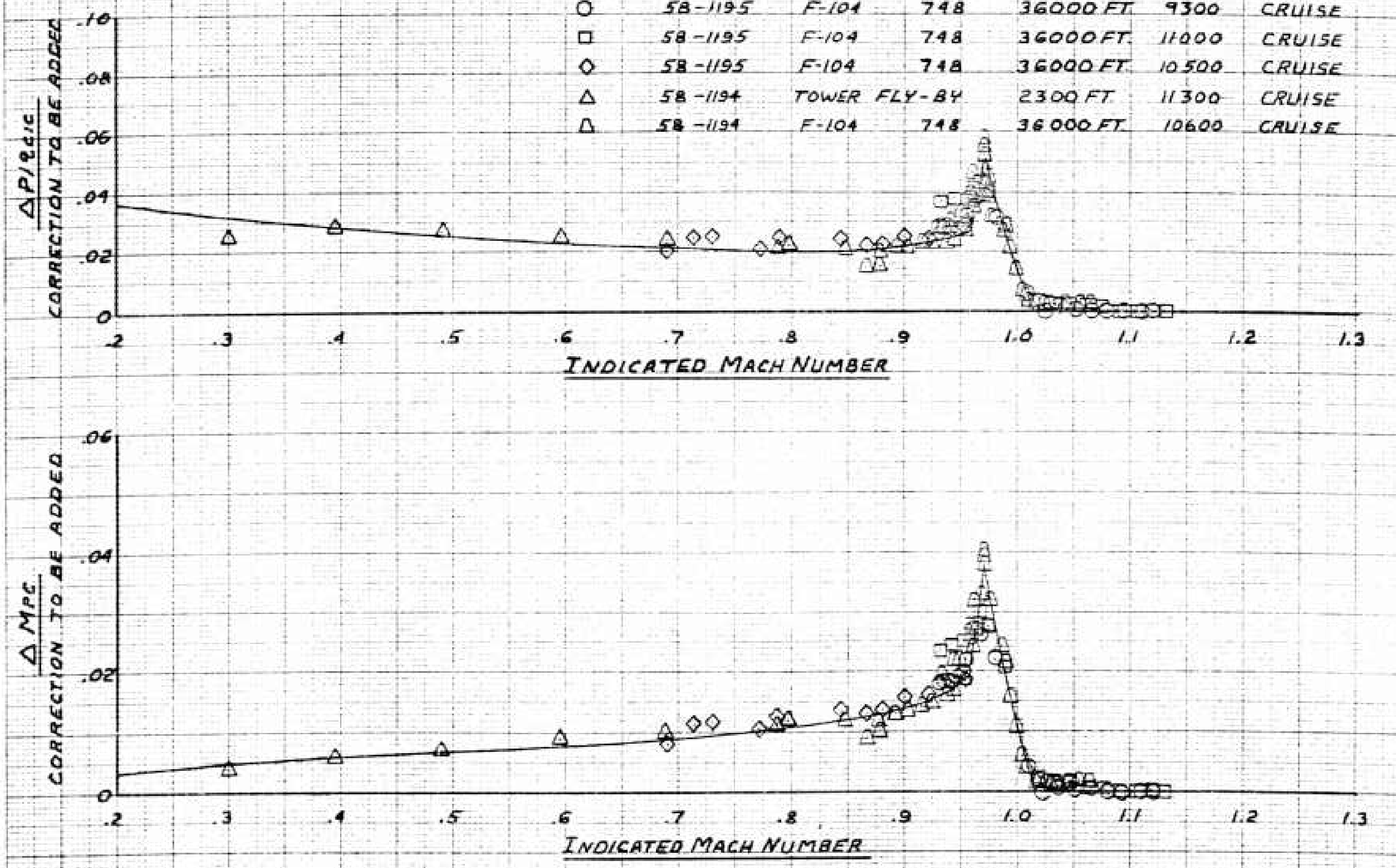


Fig. No. 74 (Cont.)

Flight Test Nose Boom Installation

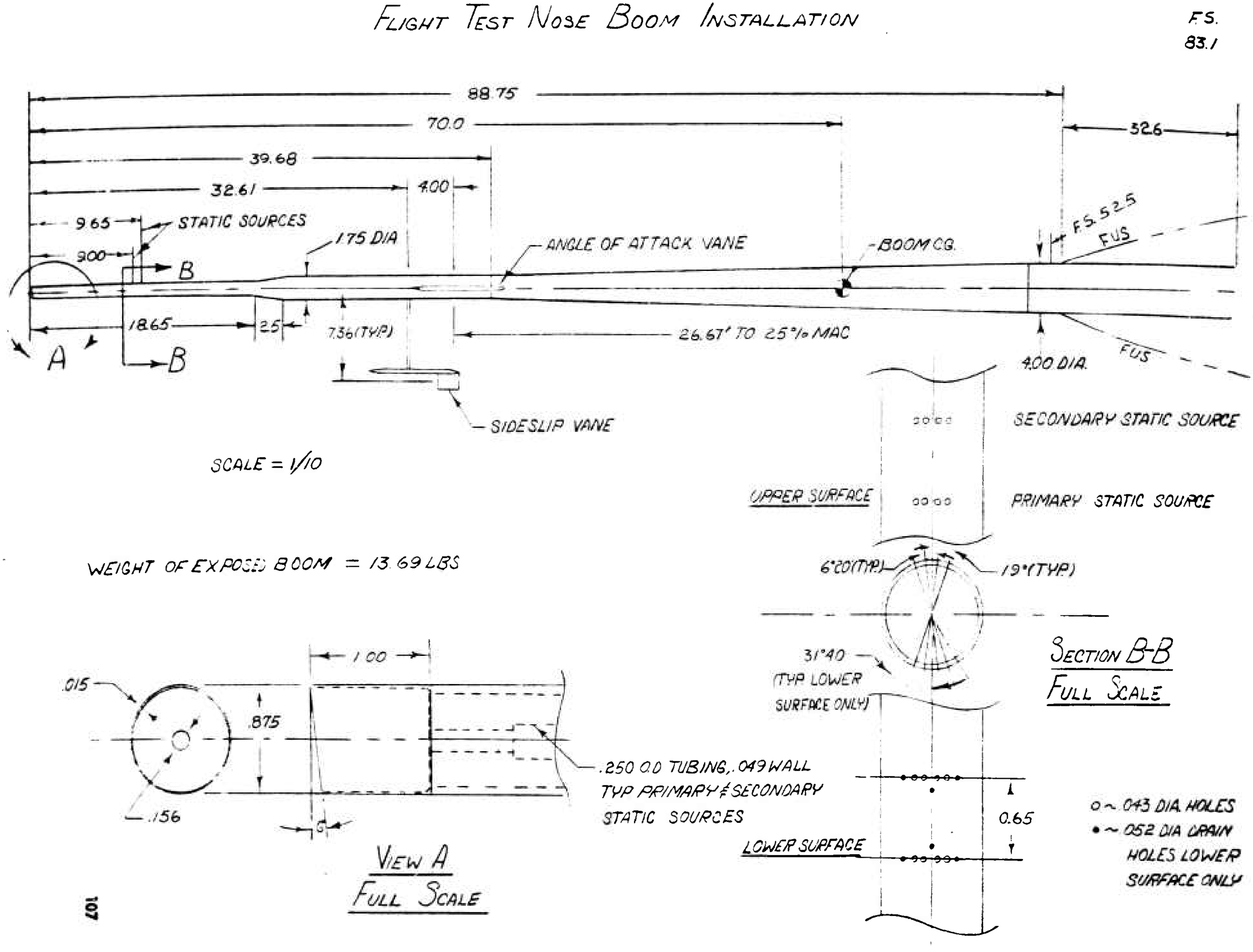


FIG. NO. 75

RUNAWAY TRIM FORCES

T-38A SN 58-1195

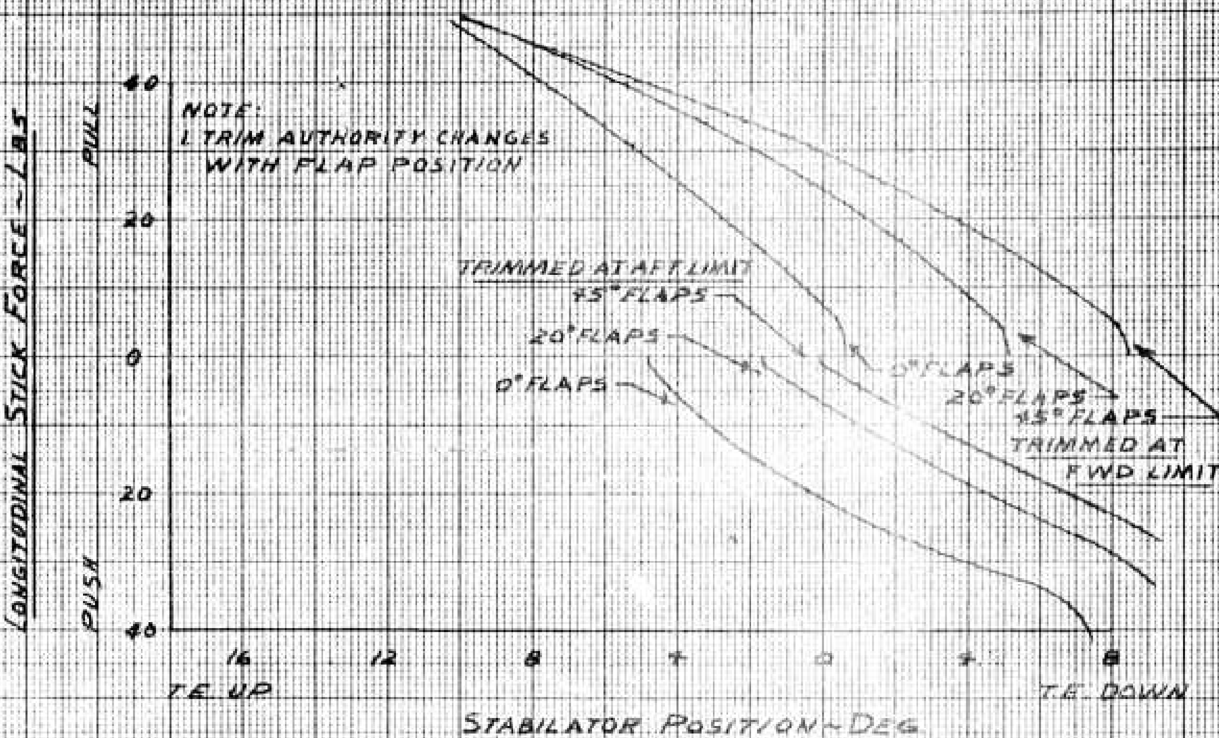


FIG. NO. 76

FLAP AUTO-TRIM SCHEDULE

T-38A SN 58-1195

APPENDIX II

general aircraft information

DIMENSION AND DESIGN DATA

Wing:

Area, total	170.00 ft^2
Taper ratio	.20
Aspect ratio	3.75
Span/thickness	51.1
Airfoil	NACA 65A004.8 Modified (.65) 50 camber
Span	25.25 ft

Horizontal Tail:

Area, total	59.00 ft^2
Area, exposed	33.34 ft^2
Taper ratio, exposed	.33
Aspect ratio, exposed	2.82
Span/thickness, exposed	58.6
Airfoil	NACA 65A004

Vertical Tail:

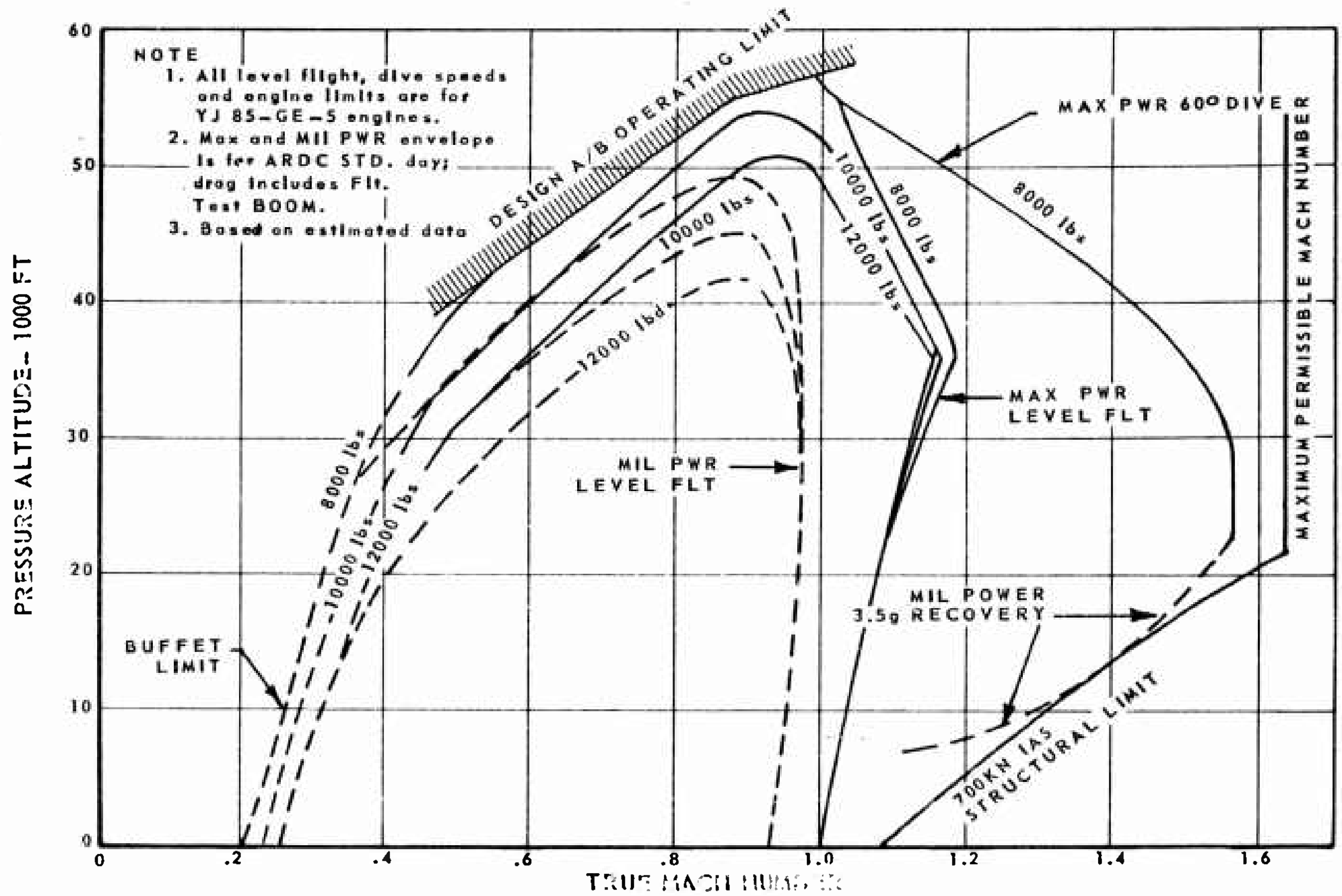
Area, total	41.42 ft^2
Area, exposed	41.07 ft^2
Taper ratio, exposed	.25
Aspect ratio, exposed	1.21
Span/thickness	42.2
Airfoil	NACA 65A004 Modified

FLIGHT AND OPERATION LIMITATIONS (as of 26 January 1961)

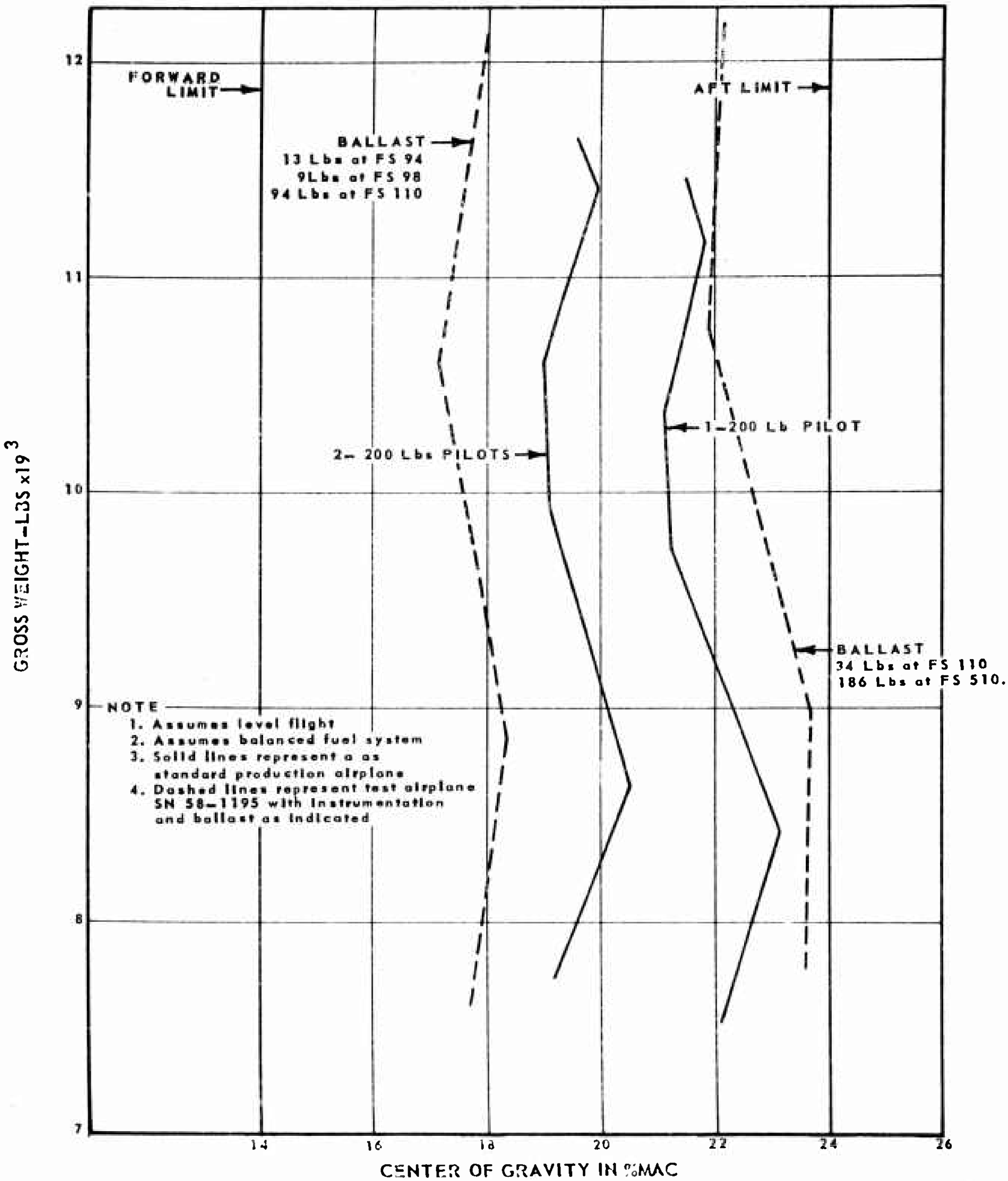
Airspeed Limitations:

Cruise configuration with speed brake extended or retracted	700 KIAS
Flaps 0 to 45 percent extended	300 KIAS
Flaps 46 to 100 percent extended	220 KIAS
Landing gear extension	240 KIAS

T-38 FLIGHT ENVELOPE
YJ85-GE-5 ENGINE



GROSS WEIGHT vs C.G.
TYPICAL CENTER OF GRAVITY DIAGRAM



APPENDIX II

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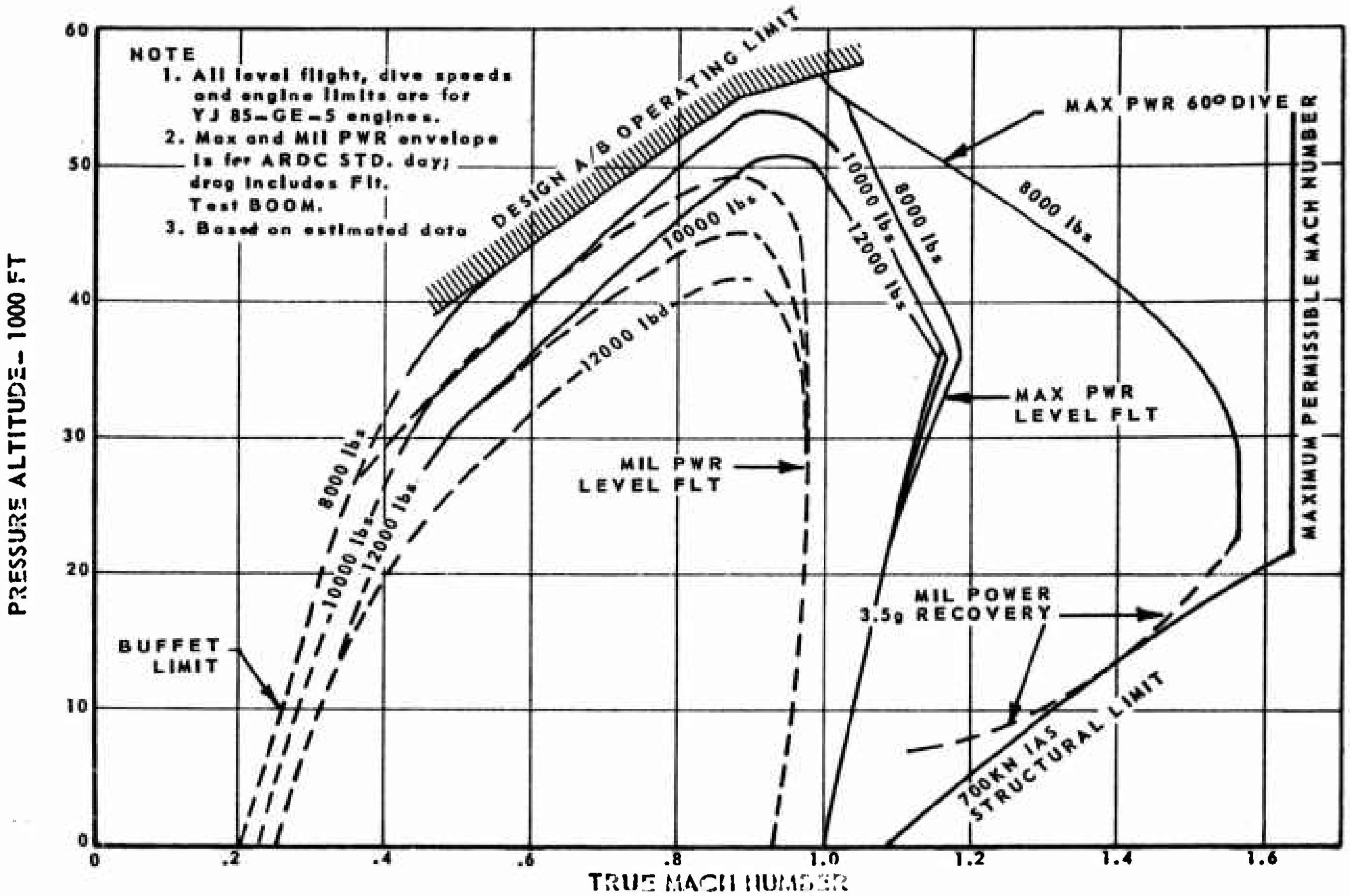
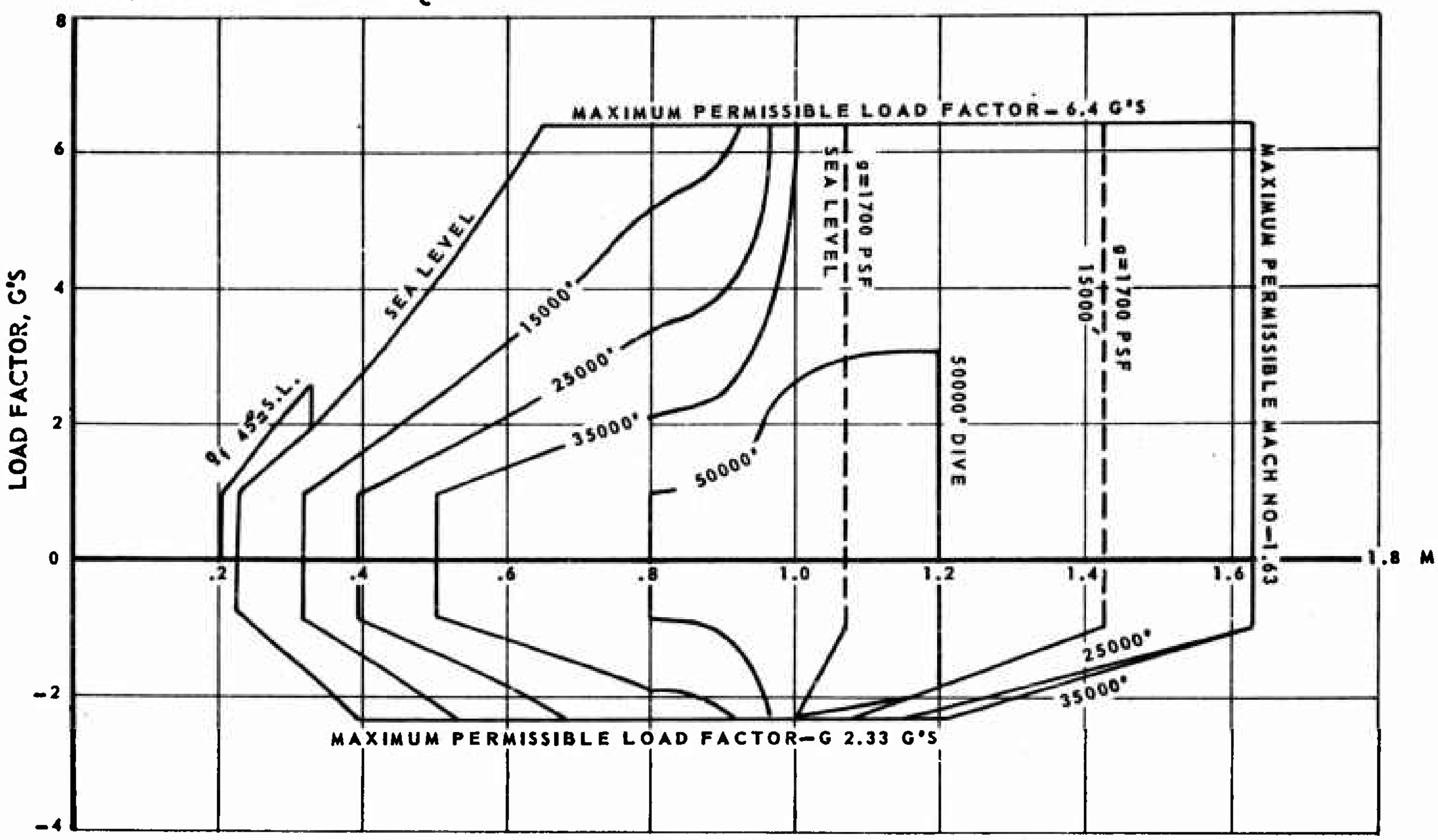
Flaps 0 to 45 percent extended	300 KIAS
Flaps 46 to 100 percent extended	220 KIAS
Landing gear extension	240 KIAS

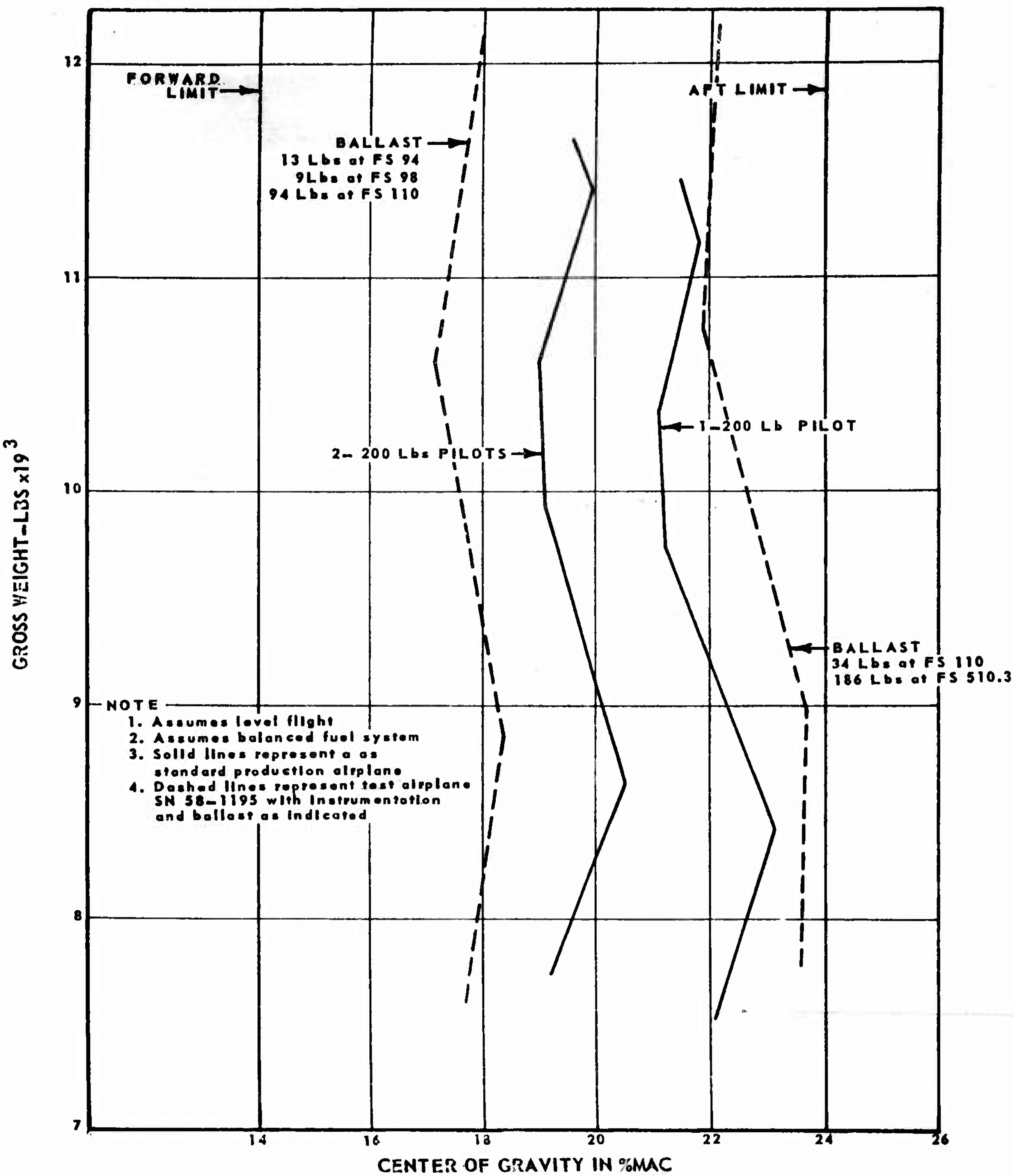
T-38 FLIGHT ENVELOPE
YJ85-GE-5 ENGINES

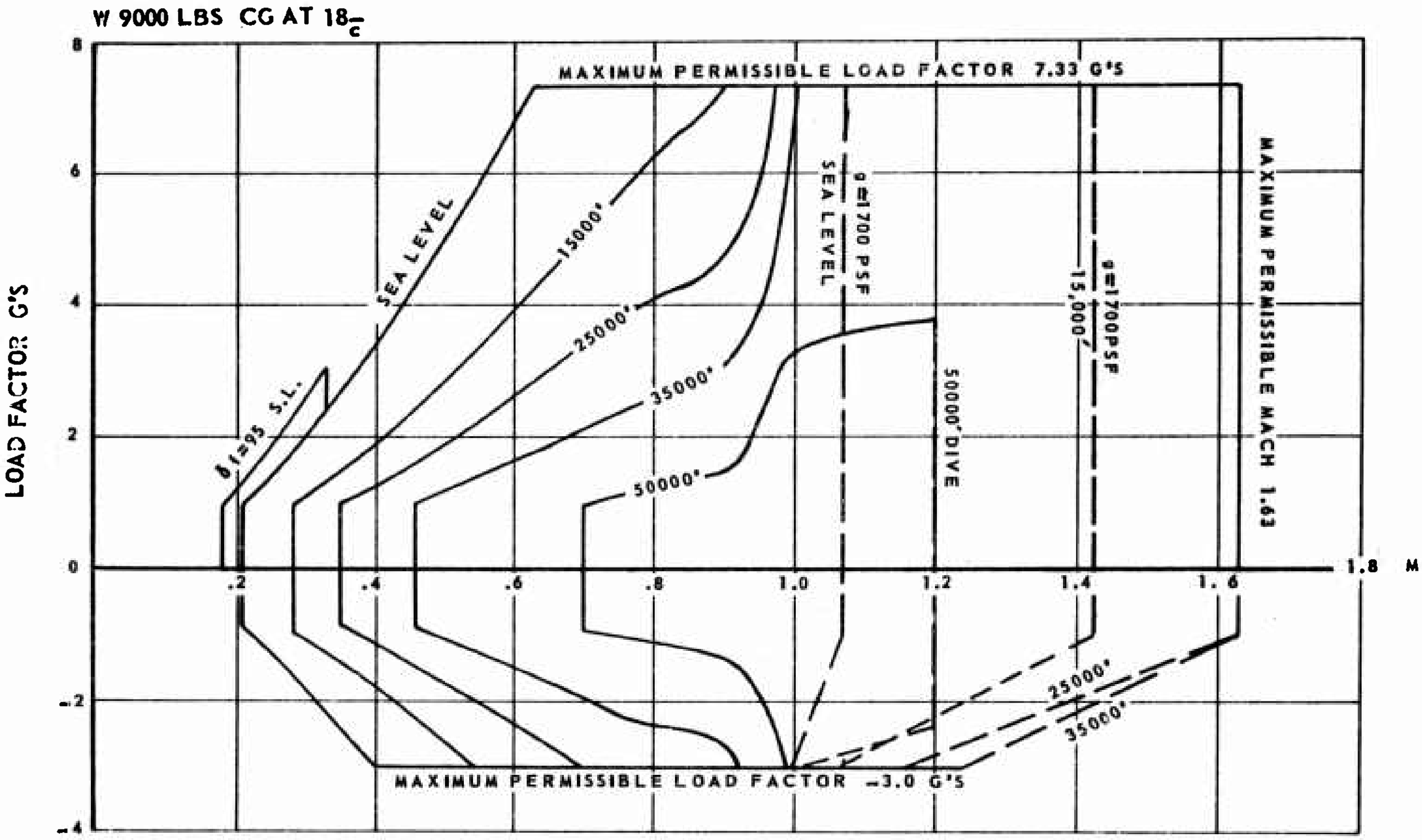


OPERATIONAL FLIGHT ENVELOPE
FULL FUEL LOADING
W=11,000 LBS C.G. AT .20$\bar{c}$



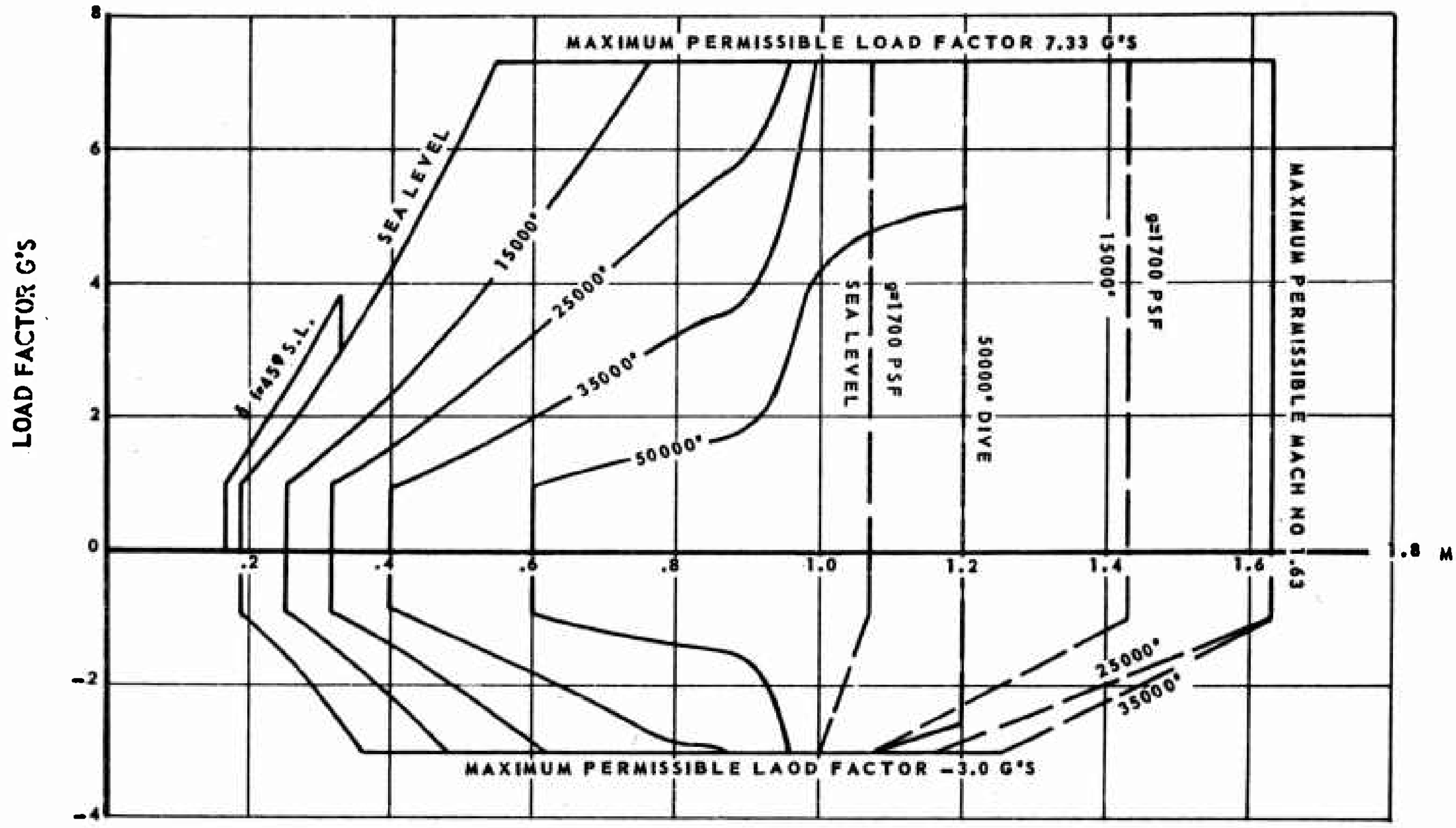
GROSS WEIGHT vs C.G.
TYPICAL CENTER OF GRAVITY DIAGRAM





W 9000 LBS CG AT 18$\bar{c}$
MAXIMUM PERMISSIBLE LOAD FACTOR 7.33 G'S
MAXIMUM PERMISSIBLE LOAD FACTOR -3.0 G'S
MAXIMUM PERMISSIBLE MACH 1.63
LOAD FACTOR G'S
SEA LEVEL
15000'
25000'
35000'
50000'
δ_f=95 S.L.
q=1700 PSF SEA LEVEL
q=1700PSF 15,000'
50000' DIVE
M

W=7400LBS C.G. AT .23$\bar{c}$



Prohibited Maneuvers:

Until the fuel system venting capability tests are complete, do not exceed:

1.5 indicated Mach number at dive angles of 0 to 50 degrees

A 50 degree dive angle for more than 5 seconds

Rolls continued past the 360 degree point.

Flight at negative load factors exceeding 10 seconds at 10,000 feet and 30 seconds at 30,000 feet (fuel starvation may occur).

Maximum speed dives with less than 650 pounds of fuel in each system (fuel starvation may occur).

Spins

The Limit Load Factor is Calculated as Follows:

$$\eta_l = 7.33 \times \frac{9600}{\text{weight}}$$

Aileron Travel Limits:

	deg	in
Full up travel (gear down)	35 $\pm$2	10.44 $\pm$0.58
Full down travel (gear down)	25 $\pm$0.5	7.51 $\pm$.15
Full up travel (gear up)	18.5 $\pm$2	5.58 $\pm$.60
Full down travel (gear up)	14 $\pm$2	4.23 $\pm$.60

Stabilator Travel Limits:

Full down (trailing edge)	8 + 1/-0	
Full up (trailing edge)	17 + 1/-0	

Rudder Travel Limits:

Left and right (gear down)	30 $\pm$1.0	14.43 $\pm$.45
Left and right (gear up)	6 $\pm$.5	3.02 $\pm$.25

Trim take-off = 0 $\pm$.4 degrees

Stabilator Trim Authority:

	Design - deg	Test (Aircraft S/N 58-1195-deg
0 degree flaps		
trailing edge down	0	5
trailing edge up	5	5

Aileron Trim Authority:

	Design - deg	Test(Aircraft S/N 58-1195-deg
Right	6	6
Left	6	5

Rudder Trim Authority:

Right	2	1.6
Left	2	2.6

Engine Ratings - Uninstalled, Standard Day, Static Sea Level:

Engine	Maximum-lb	Military-lb	RPM	EGT-°C
YJ85-1	----	2180	16,500	635
YJ85-5	3600	2450	16,500	1155
J85-5	3850	2500	16,500	1265

HYDRAULIC SYSTEM

The airplane incorporates two separate hydraulic systems. The flight control and utility systems are powered from a gear box on the right and left engine respectively. Both contain a separate reservoir which is pressurized by engine bleed air to prevent foaming or pump starvation. Both systems produce 3000 psi.

The flight control system powers the control surfaces and is outlined elsewhere in this report. The utility system assists the flight control system in this respect but also actuates the speed brakes, stability augmenter, landing gear, landing gear doors and locks, and the nose wheel steering mechanism. The gear box driving these two systems increases in ratio automatically below 68 percent engine rpm to maintain a constant pressure at low rpm. Both of these systems contain separate pressure indicators and warning devices in each cockpit.

FLIGHT CONTROL SYSTEM

The "all movable" horizontal tail, ailerons and rudder are actuated by a full powered control system. The pilot controls hydraulic servo-valves which in turn control hydraulic actuators. The airloads on the control surfaces are not transmitted to the pilot. Artificial feel,

Prohibited Maneuvers:

Until the fuel system venting capability tests are complete, do not exceed:

1.5 indicated Mach number at dive angles of 0 to 50 degrees

A 50 degree dive angle for more than 5 seconds

Rolls continued past the 360 degree point.

Flight at negative load factors exceeding 10 seconds at 10,000 feet and 30 seconds at 30,000 feet (fuel starvation may occur).

Maximum speed dives with less than 650 pounds of fuel in each system (fuel starvation may occur).

Spins

The Limit Load Factor is Calculated as Follows:

$$\eta_1 = 7.33 \times \frac{9600}{\text{weight}}$$

Aileron Travel Limits:

	deg	in
Full up travel (gear down)	35 ±2	10.44 ±0.58
Full down travel (gear down)	25 ±0.5	7.51 ±.15
Full up travel (gear up)	18.5 ±2	5.58 ±.60
Full down travel (gear up)	14 ±2	4.23 ±.60

Stabilator Travel Limits:

Full down (trailing edge)	8 + 1/-0	
Full up (trailing edge)	17 + 1/-0	

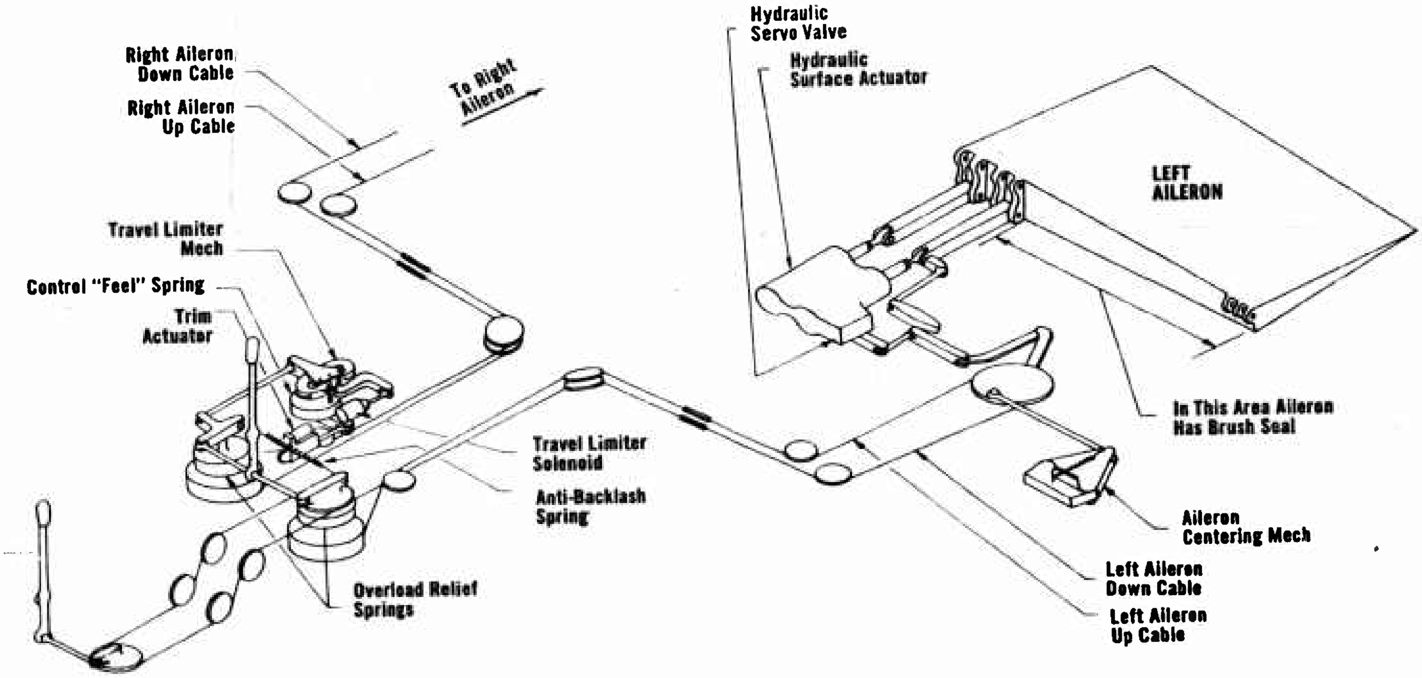
Rudder Travel Limits:

Left and right (gear down)	30 ±1.0	14.43 ±.45
Left and right (gear up)	6 ±.5	3.02 ±.25

Trim take-off = 0 ±.4 degrees

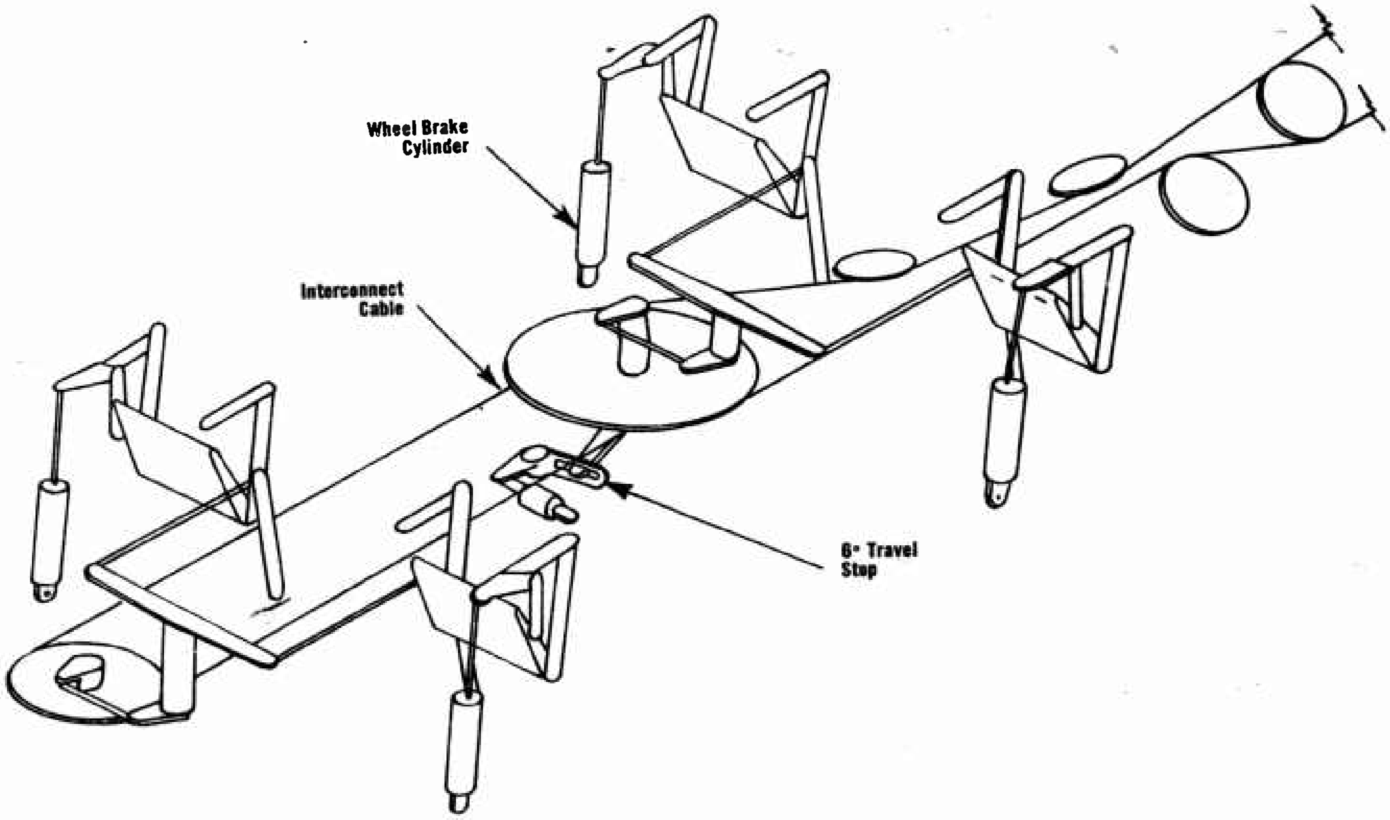
consisting of springs in the aileron and rudder system and a spring and bob-weight in the horizontal tail system, is included in the design.

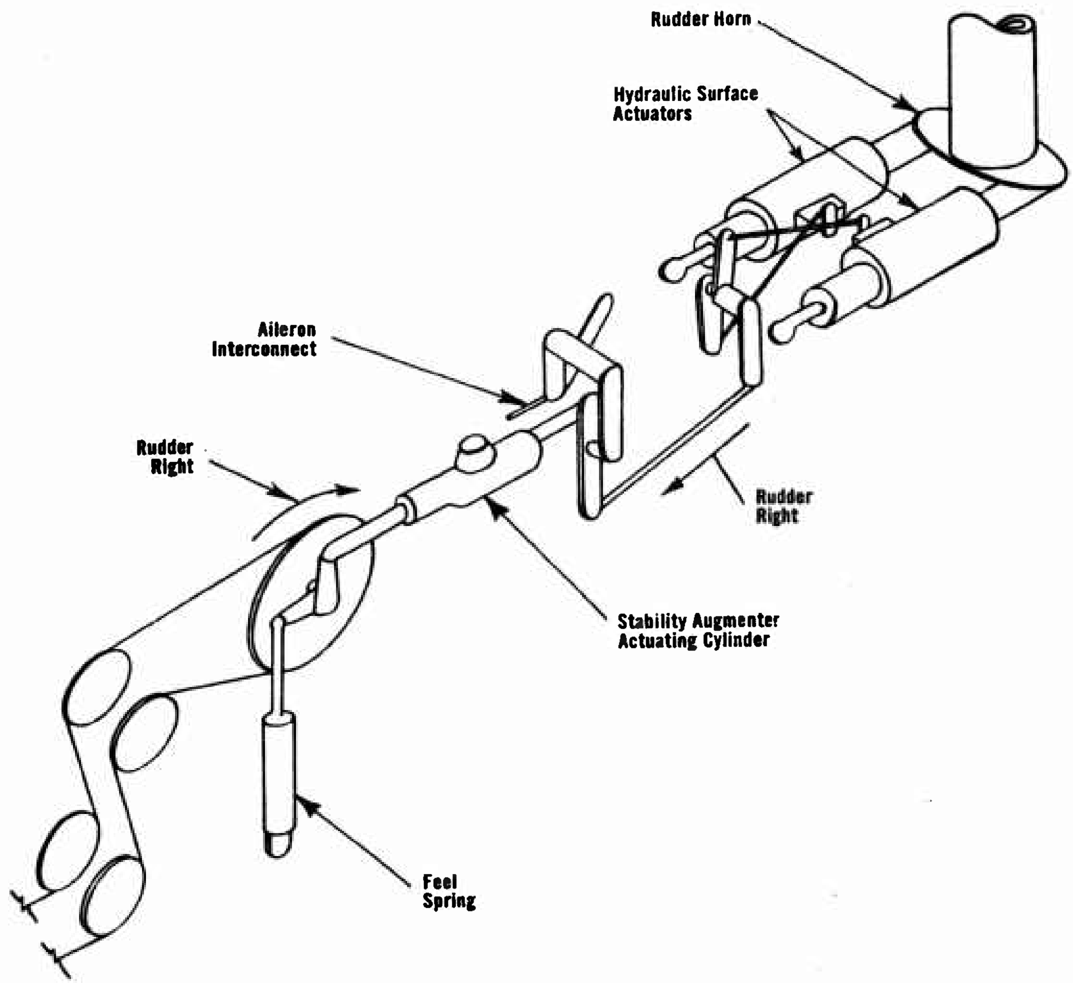
The airplane has two separate and independent hydraulic systems. Each system is powered by a variable output pump. The primary hydraulic system is powered by a pump mounted on the remote gear box and driven by a shaft from the right hand engine. This system powers the primary flight controls only. The utility hydraulic system is powered by a pump on the left remote gear box. The utility hydraulic system powers the primary flight controls and in addition the speed brakes, landing gear, gear doors, nose wheel steering and the stability augmenter system.



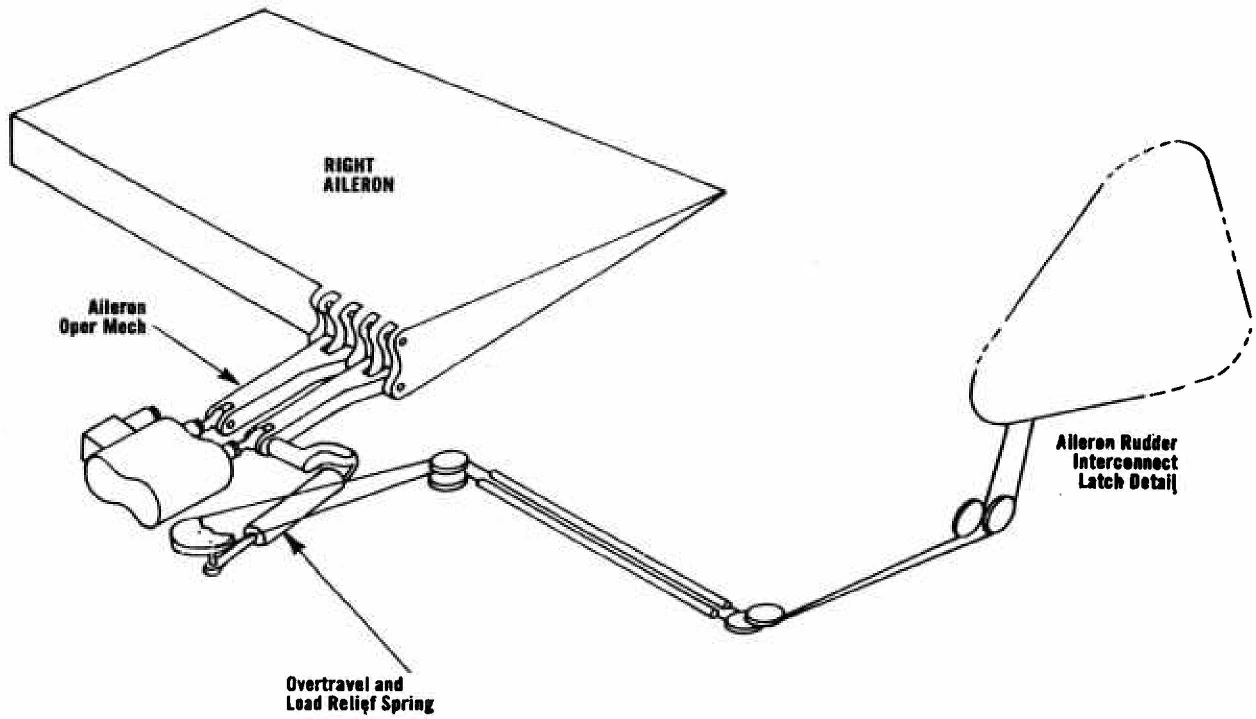
AILERON CONTROL SYSTEM SCHEMATIC
(Reference NAI 58-703)

RUDDER CONTROL SYSTEM SCHEMATIC
(Reference NAI 58-703)

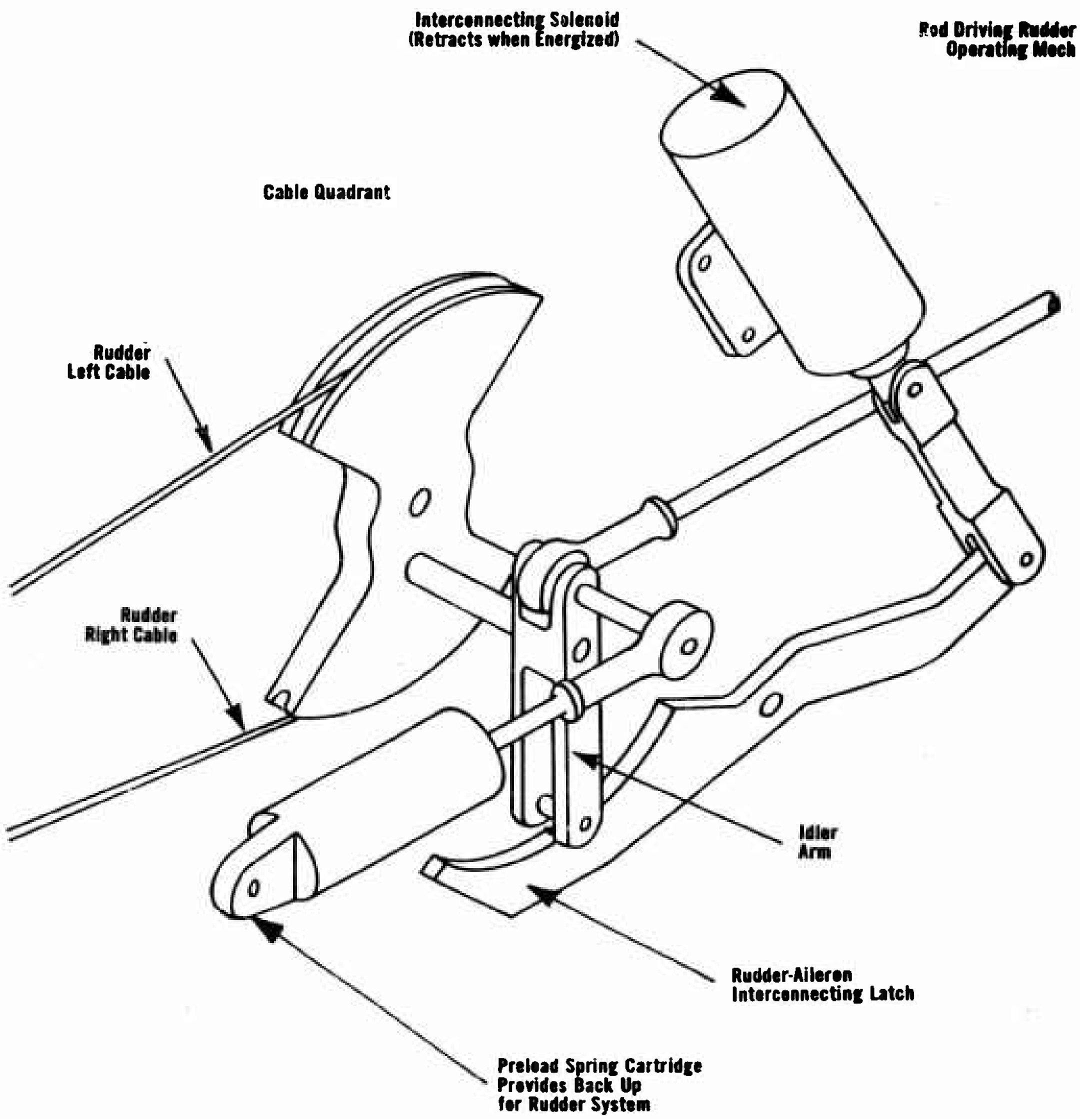



Rudder Horn
Hydraulic Surface Actuators
Aileron Interconnect
Rudder Right
Rudder Right
Stability Augmenter Actuating Cylinder
Feel Spring

AILERON-RUDDER INTERCONNECT SYSTEM



AILERON-RUDDER INTERCONNECT LATCH DETAIL



The primary flight controls are driven by two hydraulic actuators each powered by one of the hydraulic systems. A failure of either hydraulic system will result in the loss of one half the available hinge moment and, in the case of the aileron and rudder, the preload used to minimize free motion. For the horizontal tail, all the preload will be retained in case of failure of either hydraulic system since the horizontal tail is driven by two tandem actuators preloaded against each other.

In the event of flame out on both engines, windmilling rpm will supply sufficient power for the primary flight controls (at somewhat lower rates) during the descent from altitude and may be sufficient during the landing phase. Dead stick landings are not a design requirement for the airplane.

Available hinge moments for the various control surfaces are figured for 2800 psi hydraulic pressure which is a nominal value allowing for simultaneous operation of the primary controls.

The surface deflection rate is a function of the airload hinge moment. To achieve the maximum surface rates the control valves must be fully open which requires that the pilot must move the stick or rudder pedals as fast as the surfaces respond. All primary control systems have check valves such that the surfaces will not blow back until the airload hinge moment exceeds that for 3300 psi.

Aerodynamic loads on the stabilator are fully opposed by the tandem hydraulic actuators; therefore, stick forces felt by the pilot are artificially produced. The feel system consists of an electrically powered trim actuator, of the screw jack type, connected at one end to the driver of the four bar differential linkage and at the other end to a rotary mounted, spring linkage system. Safety switches located at the feel spring crank prevent adverse runaway trim by breaking the circuit to the trim actuator motor in the direction of increasing stick force; the control stick trim buttons are in the same circuit. A bobweight is mounted in the instructor's cockpit and attaches through cranks and a push-pull rod to the control stick to supplement the spring forces in proportion to load factor. A bobweight static balance spring is mounted on the torque tube in the instructor's cockpit.

The aileron surfaces are driven by dual side by side hydraulic actuators. A schematic of the aileron system is shown on page

Lateral motion of the student's or instructor's control stick is transmitted through a cable system to each aileron operating quadrant. The stick and quadrant motion open servo-valves which control the aileron actuators. A linkage termed a "follow-up" closes the servo-valve when the actuators and ailerons reach the position selected by pilot motion.

Separate small assemblies termed aileron centering mechanism are located in each wing outboard of the aileron operating mechanism, and are attached by push rod to the operating mechanism cable quadrant. The aileron centering mechanism contains a pair of springs with a combined strength great enough to return the cable quadrants, servo-valve operating linkage, and aileron surface to approximately neutral in the event of control cable failure. An identical assembly is used for both the left and right ailerons.

The aileron travel limiter is designed to restrict the aileron surface travel to 18.5 degrees up and 14 degrees down at all times when the landing gear control lever is in the retracted position. The travel limiter is operated by a solenoid and controlled by a switch in the landing gear control unit so that when the landing gear control is moved to the gear extend position the solenoid is energized and the travel limiter permits full aileron travel. An override spring is incorporated which permits the pilot to obtain up to full aileron travel (35 degrees up and 25 degrees down) with the gear retracted provided he first applies an additional override force of approximately 20 pounds at either control stick.

The rudder control system is shown schematically on page   Maximum rudder travel is ± 30 degrees. When the landing gear control lever is in the retracted position, a travel stop mechanism is engaged which limits rudder travel to ± 6 degrees. The travel stop mechanism limits the rudder pedal input only. The rudder can be deflected an additional 2 degrees by the trim system and 2 degrees input from the stability augmenter.

The surface operating hydraulic actuators are attached to airplane structure at the piston rod end. The hydraulic valves are located on the cylinders, thus motion of the cylinders provides the system follow up. The actuators are preloaded against each other to remove backlash and minimize free play in the system. The artificial feel system for the rudder is created by springs.

The aileron-rudder interconnect system ties the aileron and rudder control systems together at flight impact pressures, q_c, above 500 psf - approximately 370 knots IAS. The interconnect provides "uncoordinated" rudder; i.e., a rudder deflection opposite that usually required for a coordinated turn. At maximum gear up aileron total differential angle, $\delta_{a_{total}} = 32.5$ degrees, an "uncoordinated" rudder deflection of 7.15 degrees is produced. Schematic drawings of the system are shown on page 121

An electrical circuit containing a pressure switch is incorporated to provide the automatic interconnect signal. The right aileron motion is transmitted by cable system to a solenoid controlled coupling device which utilizes the electrical control signal to interconnect or disconnect the aileron and rudder systems. In the event of electrical or solenoid failure the interconnect will be engaged. The aileron motion signal from the coupling device is fed into the rudder surface operating mechanism through a walking beam, so that no rudder pedal motion occurs when the rudder surface is moved by the aileron signal. Conversely, rudder pedal motion by the pilot will operate the rudder surface but will not feed back to operate the aileron.

The function of the stability augmenter is to augment the inherent damping characteristics of the airplane in the longitudinal and directional modes. The rudder trim system is also incorporated into the stability augmenter system. The stability augmenter is composed of a pitch damper and a yaw damper. Short period oscillations and transients in pitch and yaw are sensed by rate gyros, one for each axis. The signals from the gyros are shaped and equalized by electronic circuitry into damping signals. The damping signals are in turn used to position servo actuators which superimpose augmenter produced surface deflections onto the pilot controlled deflections. None of the augmenter inputs can be felt on the stick or rudder pedals.

The augmenter system is powered by the left hand utility hydraulic system and the augmenter will be inoperative in the event of failure either of this system or the airplane electrical system. Under normal conditions the augmenter will be in operation throughout the flight. Either the pitch damper or the yaw damper can be turned on or off individually in flight from the front cockpit. A pressure switch turns off the stability augmenter if the hydraulic pressure drops below 1500 psi.

The stability augmenter system incorporates airspeed compensation to produce gain changes with changes in calibrated airspeed. The airspeed compensation is achieved by means of the "q_c compensator" which consists of a bellows connected to a pitot tube. The impact pressure in the bellows actuates two potentiometers, one in the yaw damper circuit and one in the pitch damper circuit. The potentiometers are linear and the circuit includes resistors to shape the signal and to produce gain decreases in both systems with increasing impact pressure up to approximately 1200 psf. The gains in both systems, as regulated by the potentiometers, are essentially constant for impact pressures above 1200 psf.

Both the pitch and yaw damper systems incorporate "wash-out" circuits which attenuate low frequency signals. In addition, low pass filter circuits act to attenuate frequencies above 10 cps. The "wash-out" circuits prevent the augmenter from opposing the pilot in steady-state maneuvers, while the low pass filters reduce the undesirable effects of high frequency noise on the system. The pitch damper authority limits are stabilator movement from + 1½ degrees (trailing edge up) to -1 degree. Failsafe circuits are incorporated in both pitch damper and yaw damper circuits which will disengage the malfunctioning channel without causing a hardover surface deflection when there is a loss of actuator feedback signal, electrical power, or hydraulic power.

The yaw damper system authority is ±4 degrees, less the rudder trim input. The trim position of the rudder is controlled from the forward cockpit by the rudder trim knob on the left console. The rudder trim acts through the yaw damper circuit and the trim limits are ±2 degrees.

WEIGHT AND BALANCE

The aircraft were weighed and the fuel tanks were calibrated. The basic weight including test instrumentation for both aircraft is listed below:

Aircraft	Gross Weight-lb	CG %MAC	Fuel gal
58-1192, basic weight	8,090	18.7	Empty tanks
58-1192, full weight	11,960	20.7	595
58-1195, basic weight	8,120	24.0	Empty tanks
58-1195, full weight	12,010	20.0	582

TEST INSTRUMENTATION

The test instrumentation was installed and maintained by the contractor. The following test instrumentation was installed in the test vehicle:

Photo Panel:

Boom altimeter
Boom airspeed indicator
Free air temperature indicator
Fuel counter (left and right)
Engine rpm gage (left and right)
Exhaust gas temperature gage (left and right)
Correlation counter
Clock

Oscillograph:

Angle of pitch
Angle of attack
Angle of sideslip
Bank angle
Normal acceleration
Longitudinal acceleration
Lateral acceleration
Stabilator position
Rudder position
Aileron position (left and right)
Lateral stick force
Longitudinal stick force
Longitudinal stick position
Rate of yaw
Rate of pitch
Rate of roll
Pitch acceleration
Yaw acceleration
Flap position
Speed brake position
Nose gear lift-off indicator
Main gear lift-off indicator

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Edwards AFB, California

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The stability and control characteristics of the aircraft are satisfactory throughout the flight envelope. Most of the undesirable features noted during the Category I tests have been improved or corrected. The longitudinal control response is sensitive at high speeds and/or Mach numbers and slow at airspeeds below 220 knots IAS, but it is the best balance available from the simple non-q-biased flight control system. The air-

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